Solar Probe

Mission and Project Description

SOLAR PROBE
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1. Introduction

This document provides background information about the Solar Probe mission and pointers to the present body of relevant scientific knowledge. This information is to be used in conjunction with Appendices A, D, E, and F by proposers in preparing a formal response to the Outer Planets Program Announcement of Opportunity.

This document contains general information, requirements, technical descriptions, and performance and interface envelopes that are pertinent to the preparation of proposals in response to the Solar Probe part of the AO. Also given is a detailed description of the activities for which the selected Principal Investigators (PIs) will be responsible. Information from the AO is repeated only if necessary for continuity of content. In the event of conflict between the provisions of the AO and this document, the AO takes precedence.

It is important to note that the reference mission described here is only one of several options under study. The AO which this document accompanies will result in the selection of Solar Probe Science Investigations, the leaders and members of which will become members of the Solar Probe Integrated Implementation Team.

The science investigations proposed by the winning teams, as well as the reference mission described in this document, will evolve together into an end-to-end mission that best meets the science objectives within the constraints of the program. The actual Solar Probe mission that is implemented may differ substantially from the reference mission and the details of the winning science investigation proposals.

NASA has not committed to this project, nor this reference mission, nor to any specific launch schedule, launch vehicle, power system, Project budget, or funding profile. In addition, the spacecraft design depicted in this AO is a conceptual, strawman design. It is likely to change during the spacecraft and payload definition phase (prior to Science Confirmation) when science and engineering teams can interact and best meet the science objectives within the constraints of the program.

The word "mission" means the Solar Probe mission. "Spacecraft" includes all launched engineering hardware and software. The term "flight system" includes all launched hardware and software for both engineering and science functions. The term "Solar Probe" may be used to refer either to the spacecraft itself or to the Solar Probe mission (as in "...will be developed for Solar Probe"). The word "project" is used in this document to refer to the Outer Planets/Solar Probe Project; Solar Probe is one of the three missions assigned to this Project.

2. Overview

2.1 Science Objectives

The science objectives and a strawman payload for the Solar Probe mission were developed by the NASA Solar Probe Science Definition Team. Their complete report can be accessed through Internet URL http://www.jpl.nasa.gov/ice_fire/SP_SDT_Report.htm.

2.1.1 Mission Overview

The Solar Probe mission will deliver a spacecraft to explore the source of the solar wind from inside the solar corona at 4 to 107 solar radii (0.5AU) from the Sun and to understand the processes that heat the solar corona and produce the solar wind.

The reference mission calls for a February 2007 launch of a single spacecraft using a Jupiter gravity assist to achieve a solar polar orbit. The probe makes two close flybys of the Sun with the perihelion at 3 solar radii from the photosphere, one flyby at or near solar maximum in October 2010, and one in the descending phase of the solar cycle about 4.3 years later. The operations for each of the two encounters consists of two phases: (1) Near Encounter: -1 day to +1 day from closest approach (radius of 4 to 20 R_S); and (2) Inner Heliosphere: -10 days to -1 day and +1 day to +10 days from closest approach (107 to 20 R_S). Instrument turn-on, checkout, calibration, and cross calibration with existing near-Earth instruments will be performed as soon as possible after launch near 1 AU. A second checkout and calibration period will be allowed just prior to the Primary Science Data Acquisition for perihelion-1 and another calibration before perihelion-2 (see Table 4).

2.1.2 Science Objectives

The Solar Probe Science Definition Team carefully considered the range of science objectives appropriate to a first close flyby of the Sun. These were then prioritized, edited for use in this AO, and their final ranking appears below. Group 1 objectives are considered to have the highest priority for Solar Probe; Group 2 objectives are considered important but not of the highest priority; Group 3 objectives are considered desirable but secondary.

The groupings resulted in a scientifically compelling set of focused goals for the Solar Probe mission:

Group 1 Objectives

- Determine the acceleration processes and find the source regions of the fast and slow solar wind at maximum and minimum solar activity;
- Locate the source and trace the flow of energy that heats the corona;

- Construct the three-dimensional coronal density configuration from pole to pole and determine the subsurface flow pattern, the structure of the polar magnetic field, and their relationship with the overlying corona; and
- Identify the acceleration mechanisms and locate the source regions of energetic particles, and determine the role of plasma waves and turbulence in the production of solar wind and energetic particles.

Group 2 Objectives:

- Investigate dust rings and particulates in the near-Sun environment;
- Determine the outflow of atoms from the Sun and their relationship to the solar wind; and
- Establish the relationship between remote sensing, near-Earth observations at 1 AU and plasma structures near the Sun.

Group 3 Objectives:

- Determine the role of x-ray microflares in the dynamics of the corona; and
- Probe nuclear processes near the solar surface from measurements of solar gamma rays and slow neutrons.

2.1.3 Measurement Objectives

The Solar Probe Science Definition Team recommended the following measurement objectives in order to achieve the Group 1 objectives using its strawman payload. NASA has edited these measurement objectives and intends for them to serve only as potentially useful information based on Science Definition Team studies with respect to meeting Group 1 objectives. Other techniques for achieving the Group 1 objectives may be proposed for which these measurement objectives may not be directly applicable. Such an alternative set of measurements could be made using different instrumentation than the strawman payload described in Section 2.1.4. Proposers should decide for themselves what is needed to meet the Group 1 science objectives in terms of the types of measurements and their accuracies, resolutions, etc., and justify their choices as part of their proposal.

The minimum measurement objectives are:

- Characterize the solar wind within a high-speed stream;
- Characterize the plasma in a closed coronal structure, viz., a coronal streamer, and probe the subsonic solar wind;
- Image the longitudinal structure of the white-light corona from the poles;
- Produce a high-resolution image in each available wavelength band, preferably in a high-latitude region;
- Characterize the plasma waves, turbulence and/or shocks, etc., that are causing coronal heating; and

- Determine the differences in solar wind characteristics during solar minimum and solar maximum.

Table 1 quantifies measurements that meet the Group 1 objectives.

2.1.3.1 *In Situ* Measurement Objectives

2.1.3.1.1 Ions and Electrons

In addition to protons and electrons, the distribution functions (from which key kinetic parameters can be derived) of dominant charge states of, e.g., He, C, O, Si and Fe should be measured with a time resolution of ~10 sec. Charge state spectra for these elements should also be obtained. The energy range should be as wide as possible. Near perihelion the probe thermal shield obscures the trajectories of particles coming from the direction of the Sun. This "shadow" cuts into the bulk of possible measurements of the distribution function of ions. The effect is mitigated somewhat by the aberration of solar-wind flow, depending, in turn, on the wind bulk flow speed. However, aberration does not fully solve the problem, and nadir viewing and a wide field of view are considered essential for measurements of the complete velocity distributions, which are expected to be broad and complex in the solar wind acceleration region. In order to make satisfactory measurements of the angular distribution of the ions in particular, nadir viewing may be necessary. (See Section 4 of this volume for a further discussion.)

All species, including electrons, can be expected to have anisotropic distribution functions, with suprathermal components that may reflect the nature of the heating process and that may be important for heat transfer. These features, as well as the detailed knowledge of the electromagnetic and electrostatic waves present, can provide very clear indications of the nature of the heating process well outside the main heating region.

If plasma microphysics plays an expected fundamental role in the physics of the corona, then to adequately address the first, second, and fourth Group 1 objectives, very fast, but more limited, ion measurements in addition to those provided by plasma and particle spectrometers are a measurement objective. To understand fully the physical processes of coronal ion heating and thermalization, measurements of the distribution functions on gyrofrequency time scales are a measurement objective.

2.1.3.1.2 Energetic Particles

The energetic particles measurement should be designed to detect the possible presence of suprathermal particles, which might be products of small solar flares (microflares), shock heating, or the high-energy tails of resonant wave absorption heating, and also to detect any trapped particle population in magnetically closed regions. It is also a measurement objective to determine the distribution function of electrons, protons, and perhaps one or two other species to satisfy the Group 1 objectives.

Table 1. Solar Probe science measurement objectives, 0.5 AU to 4 R_S to 0.5 AU. Two passes: The first near solar maximum and the second near solar minimum

Parameters or Quantity Measured	Sensitivity (Dynamic Range)	Spectral Range (Resolution)	Viewing	Resolution, Time Spatial Resolution at Perihelion (over poles)
Vector magnetic fields	0.1 nT (5 nT to 6 x 10^4 nT)			10 ms (10 ms) 3 km (1.5 km)
Time averaged distribution functions of H^+, $^3H^{e++}$, $^4He^{++}$, C^{+X}, 0^{+X}, Si^{+X} or Fe^{+X}	10 km s^{-1} (2 x 10^7)	0.05-10 keV/q	Nadir 10° x 10° and 135° x 300° (2.5 π sr)	1s or 300 km for H, He, e^- 10s for heavy ions 1s for electrons
Spectra of energetic particles by species: H, ^{3}He, ^{4}He, C, 0, Si, or Fe, e^-	10 cm^{-2} s^{-1} sr^{-1} keV^{-1} (10^7)	0.02<E<2 MeV/n e^-: 0.02 –1.0 MeV (ΔE/E) < 0.07	135° x 300° 20° x 20°	5s protons 30s others
Vector Electric and Vector Magnetic Oscillations	Threshold: 10^{-5} Vm^{-1} $10^{-9}(nT)^2$/Hz (10^6)	10 Hz to 150 kHz	4 π sr	1 ms: 1 μs snapshot, 1s spectral
Fast distribution functions of H^+		0.02 - to 5 keV (ΔE/E = 0.07)	90° x 300°	1 ms (4 ms)

(Table continued next page)

Table 1 (cont.)

Parameters or Quantity Measured	Sensitivity (Dynamic Range)	Spectral Range (Resolution)	Viewing	Resolution, Time Spatial Resolution at Perihelion (over poles)
High spatial resolution of atmosphere temperatures at different heights	10^2 erg cm^{-2} sr^{-1}	8Å at EUV wavelengths	Solar disc	5 arc sec < 1s exposure
Magnetic field (line of sight) velocity field	10 G (10 to 3000 G) 20 ms^{-1} (10 - 4000 ms^{-1})	8Å (if visible) 70mÅ	Solar disc	2 arc-sec or 32 km 2s
Solar corona (white light)	Signal to noise >1000	400 to 700 nm	20° - 180° from s/c - Sun line (<1°)	<1min

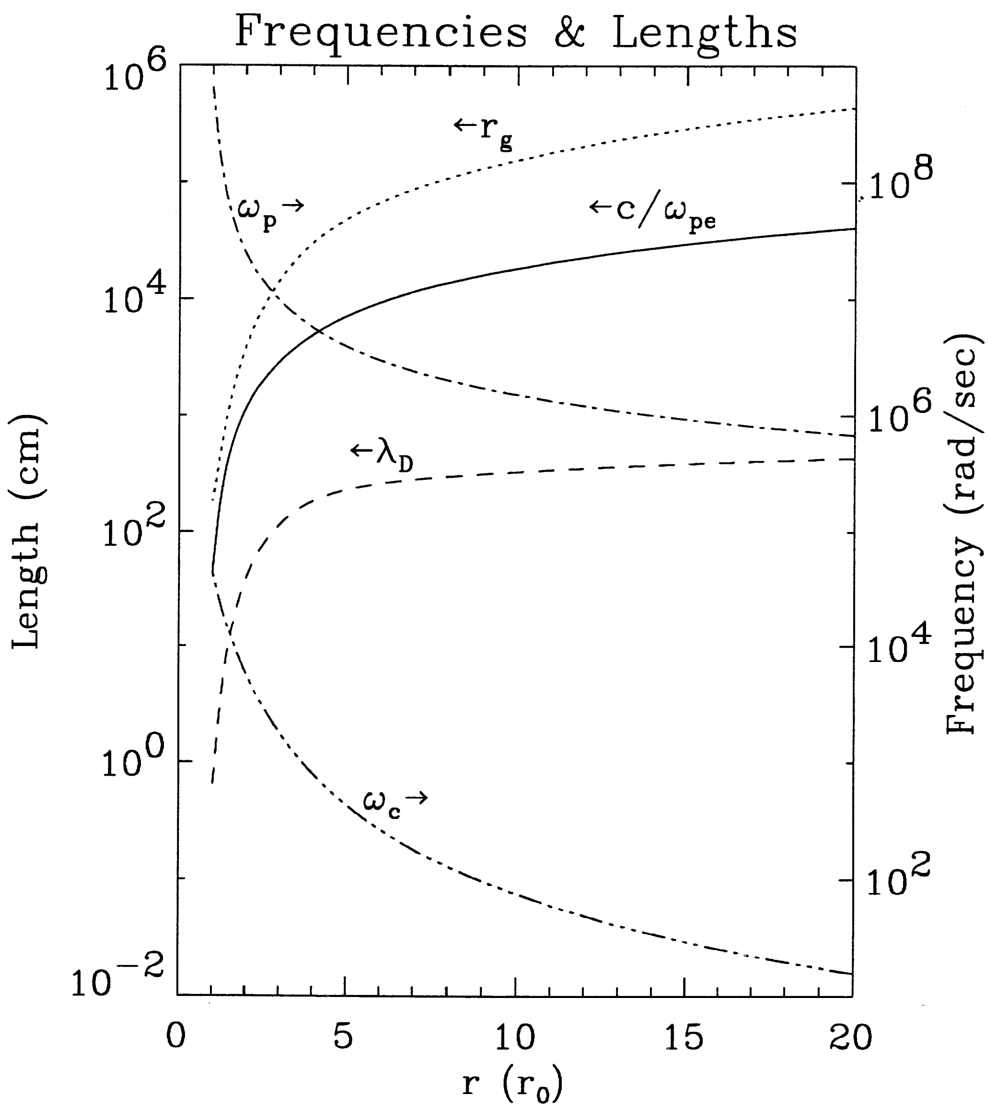

Figure 1. The variation of important plasma parameters from the solar surface up to 20 R_S according to the model of the Solar Probe Science Definition Team

2.1.3.1.3 Plasma Waves

The high field magnitudes and rest frame velocities predicted for the Solar Probe mission suggest that measurements of plasma waves and turbulence be carried out by a plasma wave system with the proper frequency and intensity coverage (see Figure 1). To achieve the measurement objectives, the sensor instrumentation need not be as sensitive as previously flown but has the objective of obtaining very high time resolution triaxial measurements (MHz). The objective is to observe ion cyclotron and low-frequency magnetohydrodynamic (MHD) waves, which may be involved in solar wind heating, whistler waves for electron thermalization processes, electrostatic emissions associated with particle beams, and shock-like structures.

2.1.3.1.4 Magnetic Fields

The large-scale solar magnetic field at 4 R_S is expected to be on the order of 0.1 G. However, due to the very dynamic nature of the corona, a measurement objective would be to measure magnetic fields as great ~0.6 G and as small as 10^{-6} G. Finally, the high speed of the spacecraft at perihelion, ~300 km/s, and the need to analyze thin structures imply a measurement objective sample rate of at least 10 vectors per sec (i.e., ~ 1 measurement per 30 km) and make the option for "burst" sample rates of ~100 vectors per second highly desirable.

2.1.3.2 Remote Sensing Measurement Objectives

2.1.3.2.1 Imaging Solar Magnetic Fields and Small-Scale Structures

A spatial resolution measurement objective in the visible and EUV or X-ray regions of the spectrum of about 20 km at the solar surface is sought in order to characterize the phenomena of interest.

In order to understand the optical design that would drive disk imaging, it helps to understand the conditions imposed by a near-parabolic orbit with closest approach at the solar equator. Table 2 contains properties for the 4 R_S closest approach case. The Table has been constructed with a measurement objective at 75 degrees latitude approach of 75 km. Because Solar Probe approaches closer to the Sun as it nears the equator, this choice allows scientists to investigate the continuum and optically thin structures with even higher resolution over most of the surface.

At 4 R_S, the measurement objective for the time between exposures is a few seconds. Blocks of characteristic images should be transmitted approximately every 10 degrees of solar latitude, but most of the data would be processed on board and the magnetic field results are transmitted rather than the images themselves. The statistical properties of the magnetic elements observed are used to determine the temporal cadence of the images transmitted.

In the EUV, a measurement objective of having 10^5 km in the field of view at 4 R_S sets the angular field of view in one aperture to be 2.7 degrees with a corresponding spatial resolution objective of 400 km at closest approach. The wavelengths of interest are the λ304 Å and λ171 Å lines. For the λ304 Å line, a second field of view is imaged with the measurement objective of achieving spatial resolution of ~30 km across a 5000 km field of view at 4 R_s.

In the soft X-ray, the measurement objective is to have a spatial resolution of ~45 km from 4 R_S. A reasonable image field of view of ~0.5 degree covering 2.5 x 10^4 km should be possible.

Table 2. Properties of a 4 solar radii closest approach mission

The rows of Table 2 show the distance from the center of the Sun to the spacecraft in R_S, the spatial resolution of the strawman telescope in km, the velocity of the spacecraft in km/s, the time to cross the equator in hrs, the velocity toward the Sun in km/s, the velocity of the point on the surface on the line connecting the center of the Sun and the spacecraft, the rotation rate of the spacecraft in degrees/hr assuming that Solar Probe is always pointed along the Sun center line, the time to move a pixel (one half a resolution element), the number of pixels in the 15 degrees latitude interval, the wavelength shift in Å, and the interferometer tilt angle required to compensate for the wavelength shift.

Property true anomaly latitude	θ=75 +75	θ=90 +90	θ=105 +75	θ=120 + 60	θ=135 + 45	θ=150 +30	θ=165 +15	θ=180 +0
Distance (Solar Radii)	10.8	8.0	6.36	5.33	4.69	4.29	4.07	4
Spatial Resolution (km)	75.0	53.6	41.0	33.2	28.2	25.2	23.0	23.5
Velocity (km/s)	188.	218.	245.	267.	285.	298.	306.	309.
Mission Time (Hours)	10.2	6.68	4.6	3.2	2.2	1.37	0.663	0
Velocity To Sun (km/s)	149	154	149	134	109	77.2	40.0	0
Velocity Surface (km/s)	10.6	19.3	30.6	43.4	56.2	67.2	74.6	77.2
Rotation Rate (deg/hr)	3.14	5.72	9.06	12.9	16.7	19.9	22.1	22.9
Time to Move One Pixel (sec)	3.54	1.39	.671	.382	.251	.187	.158	.149
Number of Pixels	5,811	7,835	9,940	11,967	13,731	15,042	15,742	-
WL Shift (Å@6302Å)	3.13	3.24	3.13	2.81	2.29	1.62	.839	0
Tilt Angle (deg@6302Å)	2.71	2.76	2.71	2.57	2.32	1.95	1.4	0

2.1.3.2.2 3-D Coronal Imager

Most of the corona is optically thin, so that an imager will see the sum of all structures in the line of sight. The white-light corona is generated from Thomson scattering of photospheric radiation from ambient free-coronal electrons. The image of the white-light corona reflects the integral of the electron density along the line of sight after appropriately accounting for geometrical factors and the variation of the ambient solar radiation along the line of sight. The solar corona has a great deal of structure, and knowledge of this structure provides information on the connection between the photosphere and the heliosphere. Plasma near the Sun is highly ionized and is, thus, in the low-beta, near-solar regions tightly bound to the coronal magnetic fields. The objective is to obtain images of these structures at predetermined intervals along the spacecraft trajectory and, by differencing techniques and tomography (assuming time stationarity), to provide Solar Probe the context of what the spacecraft has flown into/out of.

The primary measurement objectives for the white light coronal imager are: 1) obtain sufficient white-light observations during the solar encounter to create a 3-D map of the global structures of the solar corona, 2) probe the corona for filamentary structures with unprecedented resolution, and 3) obtain the first direct view of the longitudinal structure of the solar corona from the poles.

2.1.4 Strawman Payload

In December 1995, a NASA Research Announcement (NRA 95-OSS-15) was issued to solicit studies for defining and developing new and innovative concepts for particles and fields and imaging instruments that could carry out scientific investigations for a Near-Sun Flyby (aka Solar Probe or FIRE) Mission. Six concept studies were selected under this NRA:

1. "A Gated Time-of-Flight Ion and Electron Analyzer for Solar Probe"
   PI: Coplan, Michael A. / University of Maryland

2. "All-Sky and High Resolution Coronagraphs for FIRE"
   PI: Korendyke, Clarence M. / Naval Research Laboratory

3. "A Lightweight Soft X-ray Telescope for FIRE"
   PI: Krieger, Allen S. / Radiation Science, Inc.

4. "Development of Ion and Electron Plasma Instrument Including Field of View Design Concepts for the Near-Sun Flyby Mission"
   PI: Sittler, Jr., Edward C. / NASA/Goddard Space Flight Center

5. "An Innovative Imaging Package for a Near-Sun Flyby Mission"
   PI: Title, Alan M. / Lockheed Martin Advanced Technology Center

6. "An Integrated Space Physics Instrument (ISPI) for the Near-Sun Flyby Mission"
   PI: Tsurutani, Bruce T. / Jet Propulsion Laboratory

Final reports of these studies may be requested by contacting:

Mr. James E. Randolph

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Pasadena, CA 91109-8099
Phone: (818) 354-2732
Facsimile: (818) 393-9815

The Solar Probe Science Definition Team developed instrument concepts for Solar Probe that built upon the results of the above listed NRA studies. The following strawman list is an illustrative science payload used as the basis of the Science Definition Team's measurement objectives for the Solar Probe mission. Of course, none of the instruments in this illustrative set has been selected, and alternative and better approaches to meeting the Group 1 objectives may well exist.

The strawman payload described here is one conceptual design that would permit Solar Probe to meet all of the Group 1 objectives. As such it represents one possible implementation solution based upon near state-of-the-art engineering capabilities at the concept-design level. The following strawman list is comprised of two integrated instrument packages consisting of five *in situ* and three remote-sensing miniaturized instruments:

- The ***In Situ* Science** package consisting of a
  - Solar Wind Ion Composition and Electron Spectrometer;
  - Fast Solar Wind Ion Detector;
  - Plasma Wave Sensor;
  - Magnetometer; and
  - Energetic Particle Composition Spectrometer.

- The **Remote Sensing Science** package consisting of a
  - Visible Magnetograph-Helioseismograph;
  - EUV Imager; and
  - All-sky 3-D Coronagraph Imager.

Table 3 gives the estimated resource requirements for this strawman payload from the NASA Science Definition Team report (1998). The estimate for the Solar Wind Ion Composition and Electron Spectrometer includes some allowance for a nadir-viewing deflector. Note that these estimates apply only to the SDT strawman payload. The allocations for instruments proposed in response to this AO, which may be considerably different, are given in Table 7, Section 3.1.

Table 3. Solar Probe strawman payload: Instrument descriptions

Strawman Instruments	Mass (kg)*	Power (W)*	Data Rate (kbps)
Remote Sensing Instrument Package			
Visible Magnetograph-Helioseismograph	3.0	1.2	30
EUV Imager	3.0	1.2	30
All-sky, 3 - D Coronagraph Imager	2.8	2.0	2
In Situ Instrument Package			
Magnetometer (with boom cables)	1.8	0.5	1.2
Solar Wind Ion Composition and Electron Spectrometer (including nadir-viewing shield system)	4.4	4.4	15.6
Energetic Particle Composition Spectrometer	0.7	0.6	4.8
Plasma Wave Sensor (with boom cables)	3.5	2.5	9.6
Fast Solar Wind Ion Detector	1.0	1.5	19.2

* These mass and power figures represent values that use the entire allocation of mass and power. The total of proposed values, margins, and reserves must be less than the figures given here.

In this strawman, sampling techniques are mentioned only to indicate that at least one possible science implementation solution exists. The architecture and placement of the integrated packages are not described. What is provided is a guide to how the investigations could respond to the science objectives through the use of instruments with particular choices of spectral range and resolution, sensitivity and dynamic range, field of view range and angular resolutions, and time and spatial resolution. The final selection, depending on instruments proposed to the AO, may in fact be substantially different.

2.1.4.1 *In Situ* Experiments

2.1.4.1.1 Solar Wind Ion Composition and Electron Spectrometer

The strawman plasma instrument provides a study of thermal ions and electrons within the upper corona of the Sun and near solar wind. In addition to measuring the distribution functions of protons and electrons, the strawman instrument will measure the distribution functions of some dominant charge states of heavy ions. Solar Probe's heat shield obscures some particles coming from the direction of the Sun. This "shadow" at times can cut into the bulk of the ion distribution functions. This effect is mitigated somewhat by the aberration of the solar wind flow. However, aberration may not fully solve the problem. The Solar Probe

Science Definition Team agreed that nadir viewing and a wide field of view are essential to satisfy the Group 1 science objectives.

The strawman Solar Wind Ion Composition and Electron Spectrometer sensor contains a nadir viewing deflector system and two miniaturized sensors (one for nadir viewing and the other for the rest of the distribution function) and utilizes technology such as time-of-flight spectrometry, electrostatic deflection, and low-energy threshold measurements.

2.1.4.1.2 Fast Solar Wind Ion Detector

A Fast Solar Wind Ion Detector resolves ion characteristics on the scale of an ion gyroperiod in cadence with the Plasma Wave Sensors to examine the role of wave-particle effects in the acceleration (and heating) of the wind. The field of view of this instrument does not include the nadir direction. It would be mounted on the aft boom structure shown in Figure 2 yielding a 4 π-sr view minus the umbral cone (see Section 2.2.2.8). Other techniques for fast temporal responses are also possible.

2.1.4.1.3 Plasma Wave Sensor

One element of the Plasma Wave Sensor is a triaxial search coil with sufficient frequency and dynamic range to detect ion-cyclotron and MHD waves. The search coil system will also detect localized fast- and slow- mode shock waves that will also be Doppler shifted to large frequencies. These objectives require a waveform capture system. To conserve data storage resources, the waveform data could be coupled to a low-data-rate spectrum analyzer for continuous coverage and operate in either a triggered or prescheduled burst mode.

A triaxial dipole antenna is the strawman implementation for Solar Probe. Electrostatic emissions up to the plasma frequency (which will reach several megahertz at closest approach) and radio emission will be measured. This multiaxis system enables wave polarization and direction finding capabilities necessary for analysis of solitary waves (electron holes) and other electrostatic wave modes. Again, a low-data-rate spectrum analyzer can be coupled with a high-data-rate, but intermittent, waveform-sampling capability. The length of the antenna elements will be necessarily limited by the available area behind the Sun shield, thus reducing sensitivity. However, anticipated large amplitudes (potentially up to 1 V/m) will limit the need for sensitivity but, in turn, need a large dynamic range.

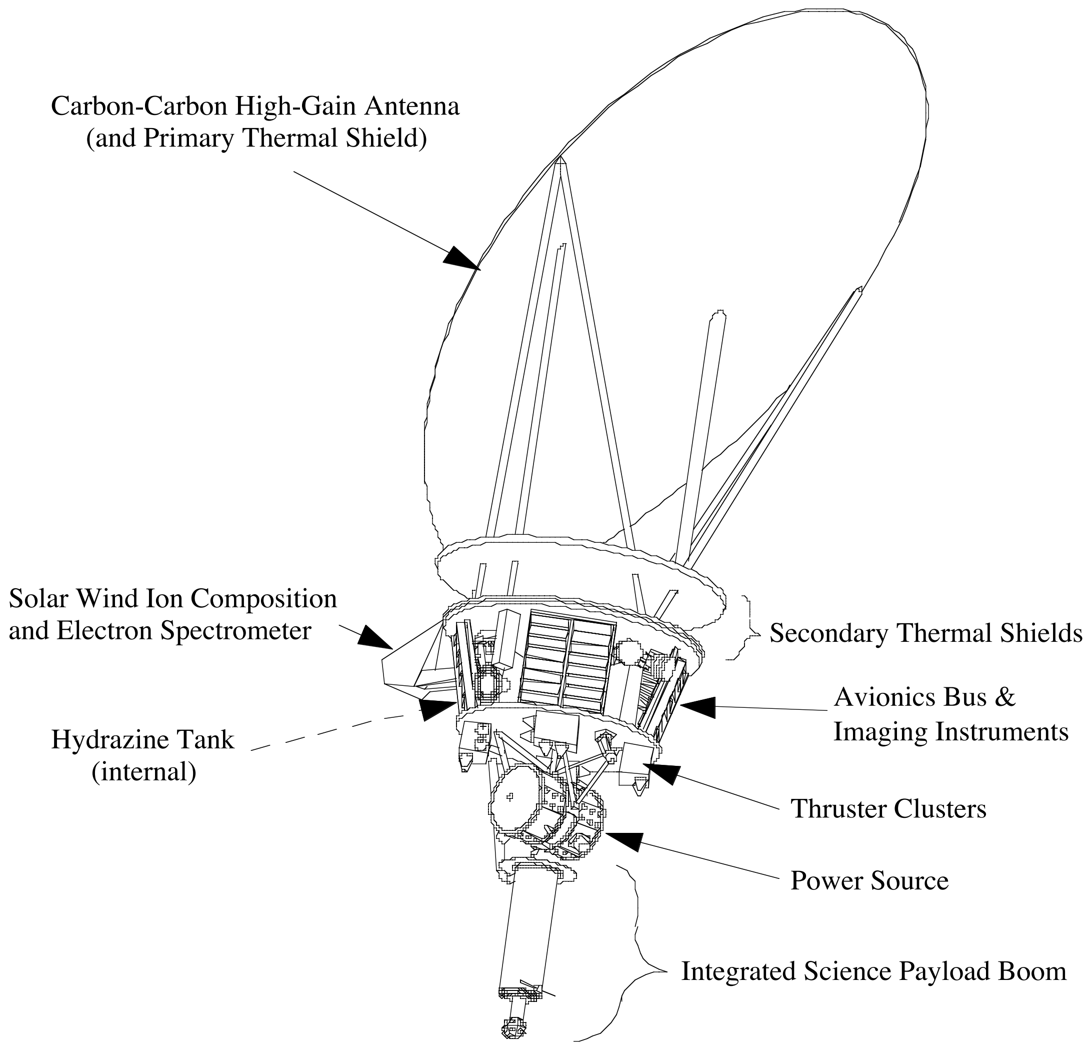

Figure 2. View of spacecraft with an example plasma instrument configuration. The electrostatic mirrors, which direct the solar wind ions and electrons within the nadir field-of-view into the spectrometer field-of-view, are shown. The first mirror is housed within the roof shaped miniature heat shield extending out from the end of a boom.

2.1.4.1.4 Magnetometer

The Magnetometer and Energetic Particle Composition Spectrometer provide links from the *in situ* measurements along the magnetic field back toward the Sun and the regions sampled by the remote sensing package. The vector magnetometers that have been used in the exploration of the heliosphere and the characterization of planetary magnetic fields are well suited to the measurement objectives of the Solar Probe mission. Coordinated measurements with the fast plasma analyzer and the plasma wave sensor are envisioned to take place at 10^2 Hz in a burst mode.

2.1.4.1.5 Energetic Particle Composition Spectrometer

The Energetic Particle Composition Spectrometer fills in the suprathermal part of the plasma distribution functions and, combined with Plasma Wave measurements, identifies accelerated particle characteristics as diagnostics for solar wind dynamics at lower altitudes than those directly sampled by Solar Probe.

2.1.4.2 Remote Sensing Experiments

2.1.4.2.1 Introduction

Although Solar Probe has unique advantages as an imaging platform, substantial technical challenges to the instrument design are presented by the high thermal loads of the close solar encounter. Other challenges occur because of the limited mass, power, and telemetry of this mission.

2.1.4.2.2 Visible Magnetograph-Helioseismograph

The most critical scientific measurements that can be made of the solar disc in the visible are, in order of priority: the magnetic field, a proxy for the magnetic field, and the continuum intensity. For the solar polar regions, the magnetic field is most likely to be clumped in isolated intergranular regions and oriented nearly vertical to the surface. Thus, Solar Probe instruments will be looking nearly straight down on the fields, so that the longitudinal Zeeman components contain most of the information. To measure the longitudinal component of the magnetic field requires spectral isolation of a portion of a magnetically sensitive line and right- and left-circular polarization analyses. In order to measure the longitudinal Zeeman effect, one wing of a Zeeman-sensitive line must be isolated. This requires a spectral bandpass of 0.1 Å.

As one example of an imager that can measure the Zeeman effect, spectral isolation can be accomplished using a solid Fabry-Perot interferometer (F-P), and polarization separation can be achieved with a polarizing beam splitter and a quarterwave plate.

The orbital trajectory presents two problems for an F-P measurement, motion blur and the Doppler shift. Both are caused by the high speed of the spacecraft along the orbital path. The exposure time for a magnetogram measurement is between 200 and 400 ms. From Table 2, motion blur is only a problem at closest approach. In the polar regions, the spectral shift is most severe. During Solar Probe’s inbound phase, the velocity component toward the Sun causes a blue shift of spectral lines. After closest approach, there is a similar motion away from the Sun, which causes a red shift.

If a direct magnetic measurement is not possible, the next most interesting indicator of the magnetic field locations is provided by images in the CH bandhead, the G-band. The bandhead is sensitive to the local heating in the flux tubes, and, thus, the intensity is a proxy indicator of magnetic field. Virtually all small bright points in the G band are coincident with compact magnetic structures.

2.1.4.2.3 EUV Imager

From Table 2, to achieve 20 km spatial resolution at the solar equator, a 2 arc-sec angular resolution telescope is necessary. In the EUV, the regions emitting light are optically thin, so in principle, arbitrarily fine structures can be observed. The desired measurements are the topology, density, temperature, and velocity of the coronal structures. Very high resolution observations of the transition region and coronal structures are a high priority for meeting Solar Probe's second Group 1 objective.

2.1.4.2.4 All-Sky, 3-D Coronagraph Imager

The more-local environment through which the probe is flying is characterized by an All-sky, 3-D Coronagraph Imager that can identify the larger structures that are being locally sampled by the *in situ* instruments. The all-sky coronagraph imager would image the ambient and surrounding corona in white light.

The All-Sky, 3-D Coronagraph Imager has 1° angular resolution and 0.1% photometric accuracy. The detector dynamic range of >1000 allows imaging of scenes with large brightness variations. Exposures are kept to <1 minute to avoid image blurring due to spacecraft motion.

2.2 Description Of Spacecraft Concept And Mission

The baseline mission and spacecraft concepts described here are for proposal purposes only. Further design work and trade-offs will be completed after science selection. The Solar Probe Mission has been designed in response to the science objectives identified by the Solar Probe Science Definition Team. What follows is the description of a "reference mission," giving a snapshot of current thinking at the time this AO was in preparation. Because the Solar Probe Mission (part of the Outer Planets/Solar Probe Project) is still in definition, many important details remain to be worked. In fact, major aspects of the entire mission architecture may be changed and improved as a result of the process in which the selected science investigation teams will become major participants. Only then will a baseline mission be determined and the design of all its elements be brought to closure and implemented. The information that follows is intended to provide proposers with a point of reference and some insight into results of developments that have taken place to date.

2.2.1 Reference Mission

The reference Solar Probe mission for this AO implements a Jupiter Gravity Assist (JGA) trajectory, which is launched in February 2007 on a Delta 3/Atlas 3-class launch vehicle

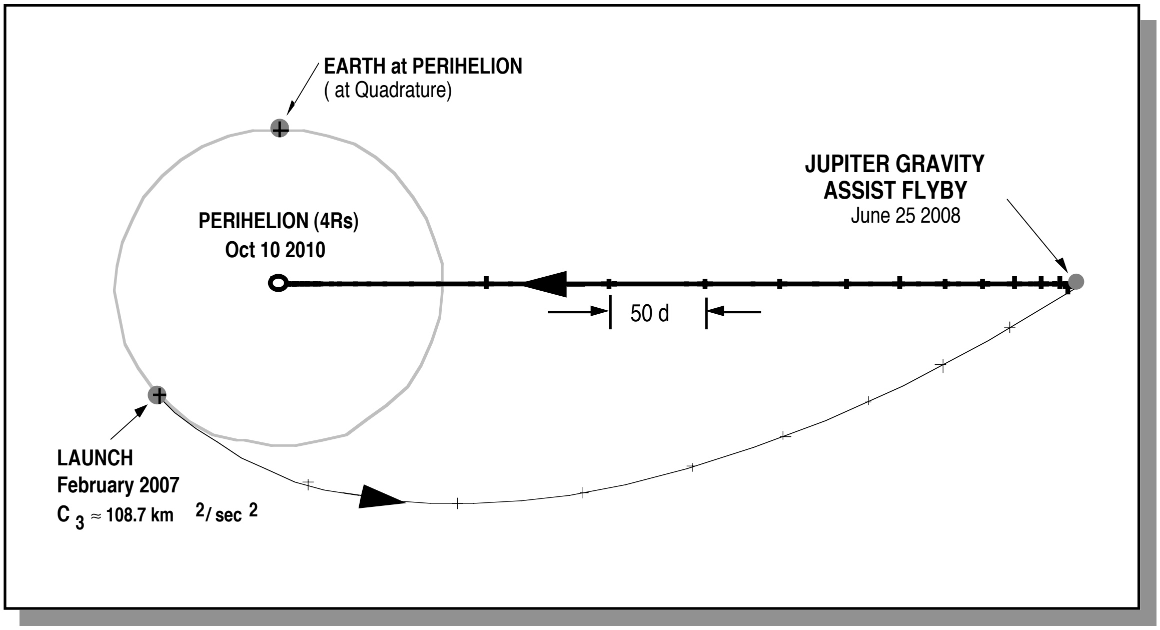

Figure 3. Interplanetary trajectory to Perihelion 1

augmented by a Star 48V upper stage. The flight duration is approximately 3.7 years to Perihelion 1, and up to 8.1 years to Perihelion 2. Figure 3 illustrates the interplanetary trajectory to Perihelion 1, and Table 4 summarizes the major events of the reference mission.

The reference mission calls for a single spacecraft launch on a Delta 3/Atlas 3-class/Star 48V launch vehicle. The interplanetary trajectory will take the spacecraft first to Jupiter, for a gravity assist, and then on to the Sun. The gravity assist flyby at Jupiter (10.5 R_J, retrograde, southern target) rotates the trajectory to a 90° ecliptic inclination and back toward the Sun for the first of two encounters with perihelia at four solar radii (4 R_S).

Recorded *in situ* and remote sensing observations of the corona and the Sun are expected to begin at 10 days before perihelion-1. At approximately 10 days prior to perihelion-1 (0.5 AU), high rate real-time telemetry will begin and continue through P+10days. The telemetry rate will vary between -40 - 50 kbps depending on the tracking station. The end of the primary observation phase for each of the two perihelia occurs at approximately 10 days past perihelion.

Table 4. JGA Reference Mission event summary

PHASE	DESCRIPTION	EVENT MARKER
Launch	Launch and Interplanetary Injection	February 15 2007
Post-launch	Instrument calibration	L + TBD d
Cruise 1	Earth to Jupiter Cruise	L+30 ^{d} to JGA-15 ^{d}
Jupiter Encounter	Jupiter Gravity Assist	June 25 2008
Cruise 2	Jupiter to P1 Cruise	JGA+15^{d} to P1-30^{d}
Encounter Preparation	Navigation and Calibration Phase	P1-30 ^{d} to P1-10 ^{d}
Start P1 Primary Mission Data Collection	Begin Primary Science Data Acquisition for P1*	P1-10^{d} (0.5 AU)
Critical Science Data Acquisition	Critical Science Data Acquisition for P1	P1±1^{d} (± 20 R_S) (Perihelion 1 - Oct 10, 2010)
End P1 Primary Mission Data Collection	End Primary Science Data Acquisition for P1	P1+10^{d} (0.5 AU)
Post P1 Transition	Playback and Calibration	P1 + 10 ^{d} to P1 + 30 ^{d}
Cruise 3	Cruise From P1 to P2	P1+30^{d} to P2-30^{d}
Encounter Preparation	Navigation and Calibration Phase	P2-30 ^{d} to P2-10 ^{d}
Start P2 Primary Mission	Begin Primary Science Data Acquisition for P2	P2-10 ^{d} (0.5 AU)
Critical Science Data Acquisition	Critical Science Data Requisition for P2	P2± 1 ^{d} (±20 R_S) (Perihelion 2 - Jan 15,2015)
End P2 Primary Mission Data Collection	End Primary Science Data Acquisition for P2	P2+10 ^{d} (0.5 AU)
Post P2 Transition	Playback and Calibration	P2 + 10 ^{d} to P2 + 30 ^{d}
EOM	End of Mission	P2 + 30 ^{d}

* High-rate telemetry not available from P1-10 ^{d} to P1-6 ^{d} and from P1+3 ^{d} to P1+10 ^{d}, and from P2 -10 ^{d} to P2 +10 ^{d}

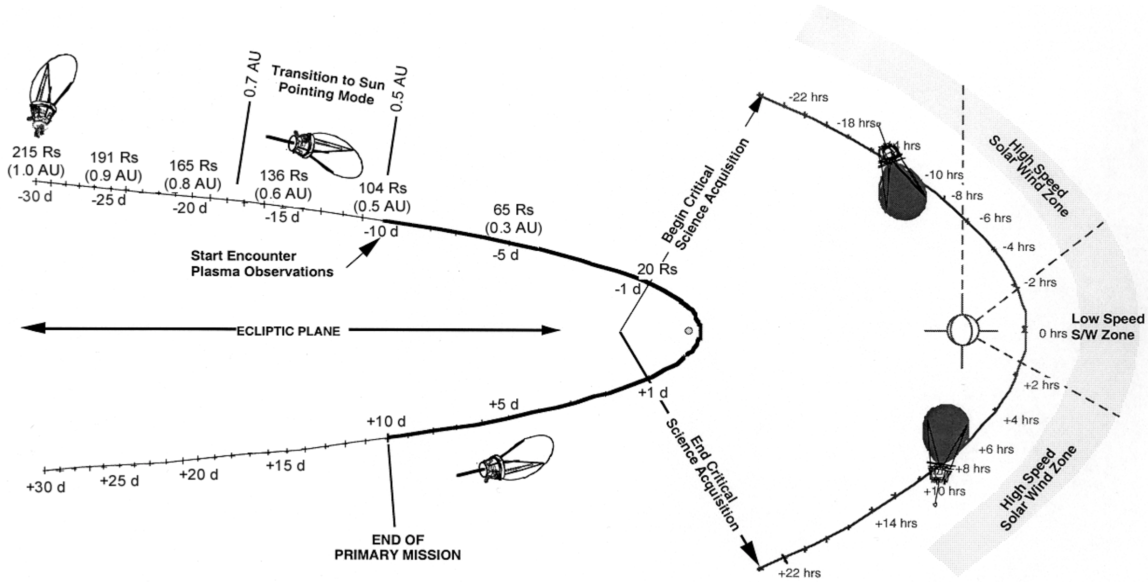

Figure 4. Perihelion 1 trajectory as seen from the Earth

During the first perihelion passage, quadrature geometry is available (see discussion below), and science data will be transmitted in real-time with high-priority data also stored onboard. The stored science data will be played back as soon as possible after the end of the critical data acquisition period (see Figure 4).

2.2.1.1 Baseline Mission

The Solar Probe trajectory uses a northern approach to the Sun reaching a speed in excess of 300 km/s at perihelion. This results in a "pole-to-pole" passage of approximately 13 hours. Views of the Sun as seen from the spacecraft at various times during approach for a field of view (FOV) of 30° are illustrated in Figure 5.

The conceptual spacecraft includes an extendible aft science boom. After launch, the boom may be extended for a calibration and then retracted and kept retracted until the spacecraft nears the Sun. At the first perihelion pass, the boom will be extended at ~0.5 AU and gradually retracted as the spacecraft nears the Sun so that the tip remains continuously just inside the sun shield umbra. The aft boom is gradually extended following the 4 R_S point so that its tip again remains continuously just inside the umbra. After the first perihelion pass is complete (>0.5 AU), the boom is retracted again. The same boom positioning scenario is repeated during the second perihelion pass.

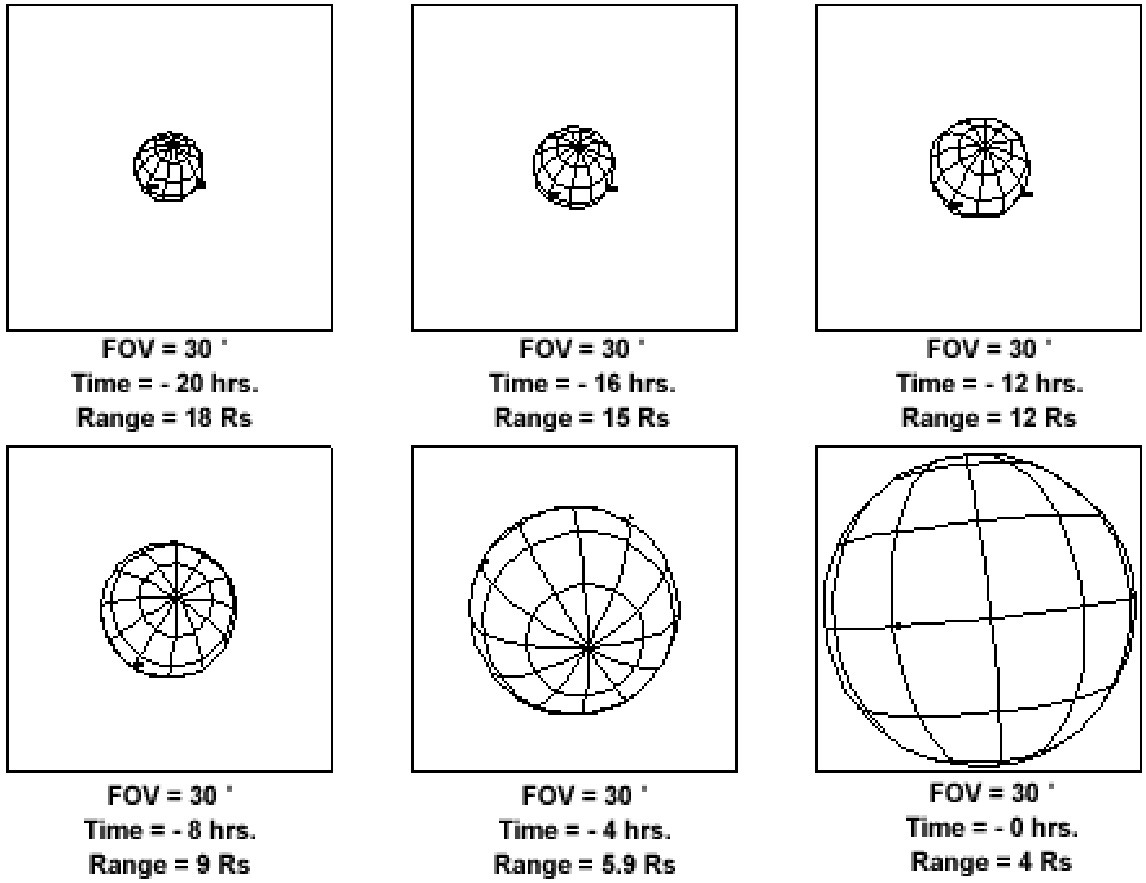

Figure 5. Typical incoming approach perspective

In the reference mission design, the time of the first perihelion is selected to allow the quadrature geometry (Sun-spacecraft-Earth angle = 90 deg.), which assures a continuous high-rate data link to Earth through the dual-purpose thermal shield/high-gain antenna of the spacecraft. The Project expects to provide such geometry at the first perihelion but cannot guarantee quadrature for the second perihelion passage. The time of perihelion is also selected such that the position of the Earth allows Earth viewing of the perihelion longitudes just prior to spacecraft overflight.

Because the high-gain antenna is body mounted, a time period will occur during the first encounter such that the direction of the Earth will be outside the pointing capabilities of the high gain antenna while maintaining the necessary shield pointing for thermal control. This time period is expected to occur from perihelion -10 days to perihelion -6 days with the aft boom fully retracted (or –4 days with the boom fully extended). During this time, the real-time downlink to the Earth will not be possible, and data storage will be necessary. Real-time data outage will also occur on the outbound leg from +3 days with the aft boom fully retracted (or +2 days with the boom fully extended) to +10 days, and again, data storage will be necessary.

Quadrature conditions are not enforced for Perihelion-2; therefore, real-time data return is not implemented for the second perihelion passage.

2.2.1.2 Orbit Determination Accuracy

Radiometric tracking alone will be used to deliver the spacecraft to its Jupiter aimpoint to an accuracy sufficient to allow subsequent delivery to the Sun to 4±0.1 R_S. Time-of-flight (downtrack) and out of plane components will be controlled only to the extent that a real-time link remains enabled at Perihelion 1. The postmission reconstruction of the trajectory will be to a few km, one sigma, in all three components, for Perihelion 1. If there is no real-time link at Perihelion 2, then the reconstruction will be degraded (to be determined) relative to that at Perihelion 1.

2.2.2 Spacecraft System Design

2.2.2.1 Applicable Standards

The following standards apply:

- The metric system of measurement;
- X2000 Mission Data System standards for software implementation; and
- Reliability, Quality Control, and Safety standards will be tailored to the mission with specific emphasis as appropriate for a long, but resource-limited, mission and in accordance with the project risk management approach.

2.2.2.2 System Overview

The flight system for the reference mission is envisioned to consist of a 3-axis stabilized spacecraft bus that houses the engineering and science electronic subsystems, a thermal shield/high-gain antenna subsystem, a propulsion subsystem, an attached proposed Advanced Radioisotope Power Source (ARPS), and an extendible aft science boom. The actual spacecraft power source is yet to be defined; however, the ARPS creates a more challenging radiation environment to which the science payload should be designed. Several views of the spacecraft concept are shown in Figure 6. The radiation environment is described in the Environmental Requirements document of the Outer Planets Program Library, available over the Internet through URL http://outerplanets.LaRC.NASA.gov/outerplanets.

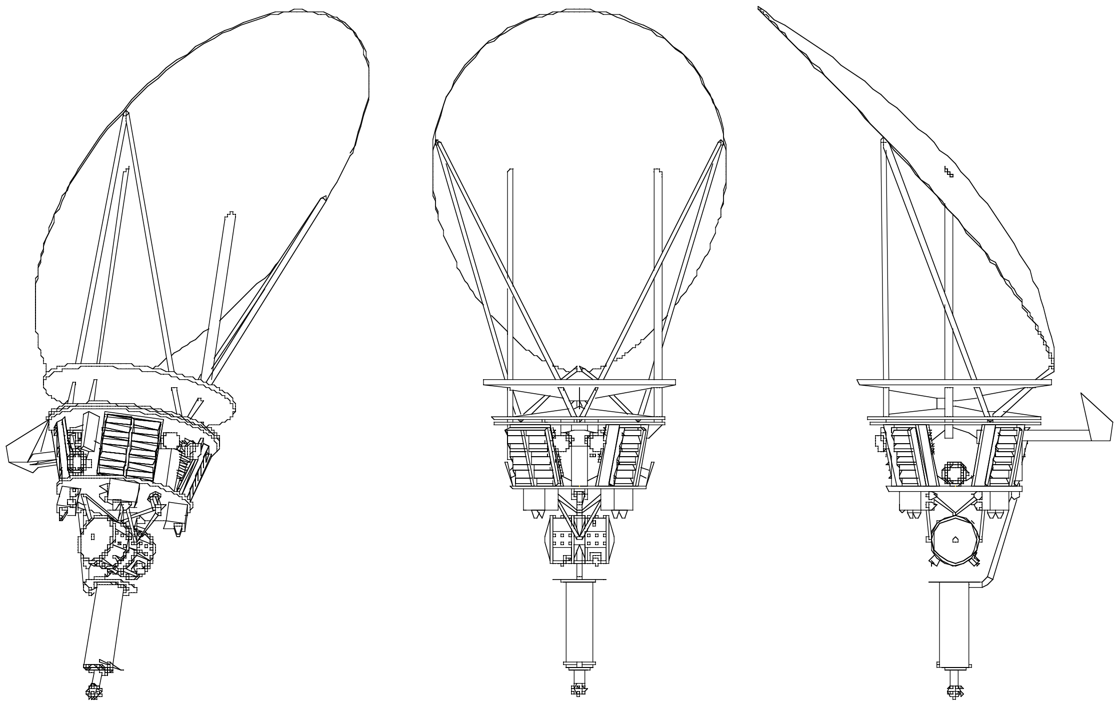

Figure 6. Solar Probe spacecraft

The major hardware elements are depicted in Figure 7.

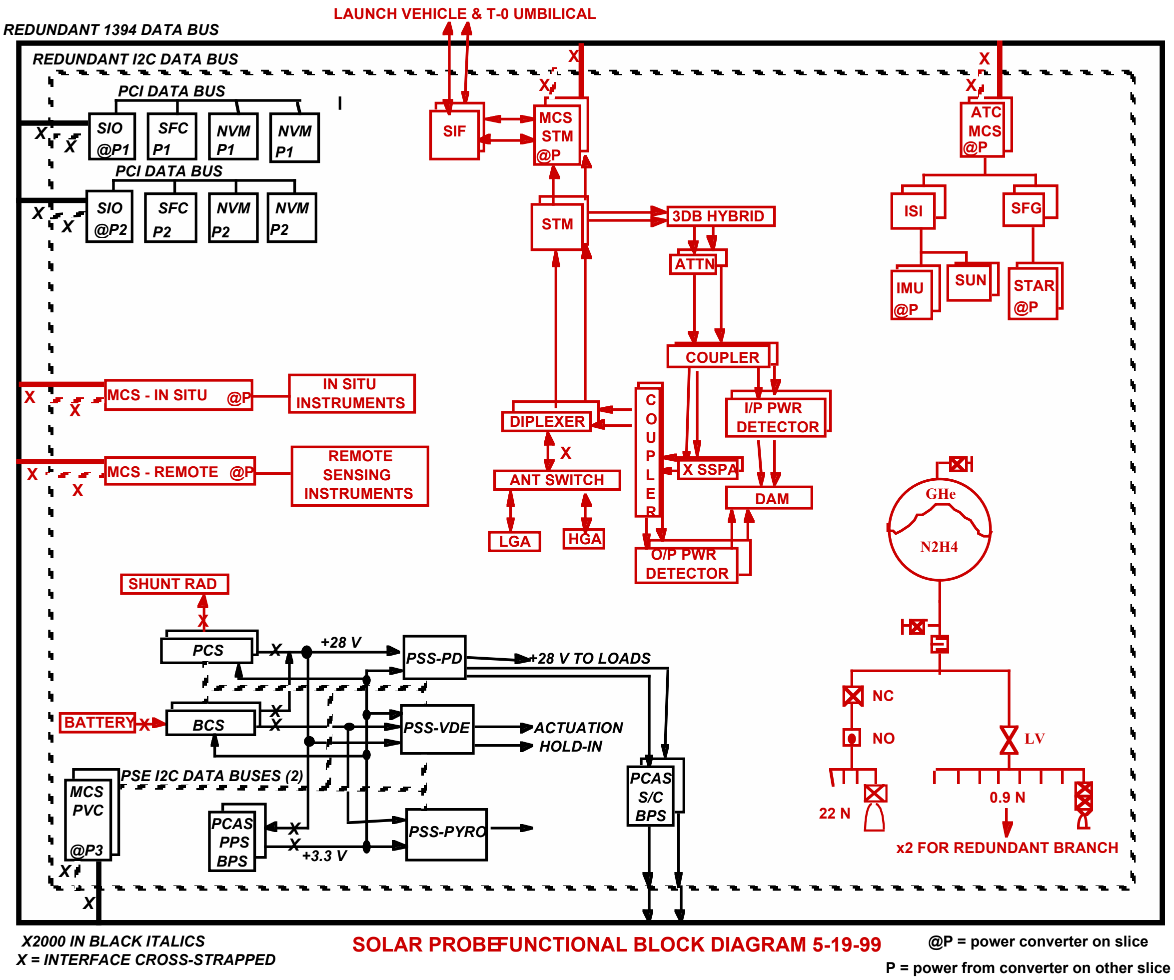

Figure 7. Solar Probe functional block diagram

The current approach assumes that a substantial portion of the engineering avionics subsystems will be designed and qualified through the JPL X2000 development program. The avionics design will incorporate advanced technologies to allow integration of several functions onto a single substrate. By decreasing the size of the electronics while increasing functionality, the avionics mass will be significantly decreased for the Solar Probe mission as compared to missions up to now. The integration of the avionics into a small volume will also reduce the mass of the cabling required to integrate these functions. Additionally, the majority of the electronics developed by X2000 will be radiation hardened to 1 Mrad, and, therefore, very little, if any, additional shielding mass will be required to meet the radiation hardening specification requirement for Solar Probe, which is currently estimated at 35 krad (Si).

ACRONYMS & ABBREVIATIONS USED IN FLIGHT SYSTEM BLOCK DIAGRAMS

ANT - ANTENNA

ARPS - ADVANCED RADIOISOTOPE POWER SOURCE

ATC - ACS CONTROLLER

ATTN - ATTENUATOR

BCS - BATTERY CONTROL SLICE

BPS - DATA BUS POWER SLICE

CX - COAX

GHe - GASEOUS HELIUM

HGA - HIGH GAIN ANTENNA

I2C - DATA BUS STANDARD

IMU - INERTIAL MEASUREMENT UNIT

ISI - IMU/SUN SENSOR INTERFACE SLICE

LGA - LOW GAIN ANTENNA

LV - LATCH VALVE

MCS - MICROCONTROLLER SLICE

MGA - MEDIUM GAIN ANTENNA

N - NEWTON

NC - NORMALLY CLOSED PYROVALVE

NO - NORMALLY OPEN PYROVALVE

NTO - NITROGEN TETROXIDE

NVM - NONVOLATILE MEMORY

N2H4 - HYDRAZINE

PCAS - POWER CONVERTER ASSEMBLY SLICE

PCI - DATA BUS STANDARD

PCS - POWER CONTROL SLICE

PCAS - POWER CONVERTER ASSEMBLY SLICE

PD - POWER DISTRIBUTION

PSE - POWER SUBSYSTEM ELECTRONICS

PSS - POWER SWITCH SLICE

PVC - POWER/PDE/VDE MICROCONTROLLER

PYRO - PYRO DRIVE ELECTRONICS

PWS - PLASMA WAVE SPECTROMETER

REG - REGULATOR

RS422 - DATA BUS STANDARD

RW - REACTION WHEEL

RWE - REACTION WHEEL ELECTRONICS

S/C - SPACECRAFT

SFG - STELLAR FRAME GRABBER

SFC - SYSTEM FLIGHT COMPUTER

SIF - STM INTERFACE SLICE

SIO - SYSTEM INPUT/OUTPUT INTERFACE

SRU - STELLAR REFERENCE UNIT

STAR - STELLAR REFERENCE UNIT

STM - SPACE TRANSPONDING MODEM

SUN - SUN SENSOR

VDE - VALVE DRIVE ELECTRONICS

WG - WAVEGUIDE

X - CROSS STRAPPED INTERFACE

X SSPA - X-BAND SOLID STATE POWER AMPLIFIER

Since X2000 is just getting started and has a very aggressive program, some of their deliverable products may not have the performance envisioned today. Whenever possible, this has been foreseen in this AO by the science allocations identified. As X2000 matures and the final flight performance and components are determined, the flight system and instruments will need to review and finalize the functions and capabilities to be flown. The approach assumed for the integration of the science payload into the engineering system is to minimize the duplication of functions and, thereby, allow maximum science return for the minimum mass, power, and volume. To achieve this, an integrated team must determine the distribution of functions and requirements between the science payload and the spacecraft engineering system. Concurrent engineering and teamwork between science and spacecraft will be required throughout the design and implementation phase to ensure that cost targets and science objectives are met within the resource constraints of the mission. For the purposes of this proposal, however, the allocations of resources and functions to the science payload specified herein should be assumed.

2.2.2.3 Mass

The allocated mass for the Solar Probe science instruments is given in Table 7 (Section 3.1) including radiation shielding and contingency. The mass allocation places no constraints on where the mass is located within the volume constraints described in 2.2.2.5 below except for mass on the aft boom. Mass of approximately 1 kg maximum at the tip of the boom and 2 kg maximum nearer the center of the boom extension can be accommodated in this conceptual design. The total science mass does not include the optical baffles through the HGA/Thermal Shields supporting the nadir viewing optical instruments and the (present concept) aft instrument boom structure and actuator. Any other booms for instrument mounting must have their mass included as part of the instrument mass.

2.2.2.4 Power

The power allocated for science is given in Table 7 (Section 3.1). This allocation is a maximum for any given point in time during the mission except for possible short-term contamination prevention. The sum of the power for the science complement may exceed this number, as long as operationally the science observations are sequenced so that no more than the allocation is required at any one time. Power transients of up to 100 W for ≤50 msec are acceptable.

The power subsystem has not yet been determined for this mission. For purposes of a common reference and because the instrument environments would be the most challenging technically, a radioisotope power source is considered here.

The Power Subsystem regulates and converts the output voltage of the ARPS such that loads receive regulated power between 22 and 36 VDC. Providing other regulated voltage levels and any high-voltage requirements will be the responsibility of the science investigation. Each switched power line will have associated telemetry reporting on/off status, trip status, current level, and output voltage.

2.2.2.5 Volume

The volume accommodations for science instruments are broken into two sets: 1) bus mounted instruments, and 2) instruments mounted on the aft boom. Since all hardware on the Solar Probe Spacecraft must be contained in a small conical umbra and shaded from the intense solar flux, volume is a critical resource for this mission.

The volume allocated to the bus mounted instruments is subdivided into 3 equally sized tapered wedges, as shown in Figure 8. The three 90° wedges are located on the +Y, -X and –Y axes. No margin exists in these volumes; therefore, the instruments must carry volume

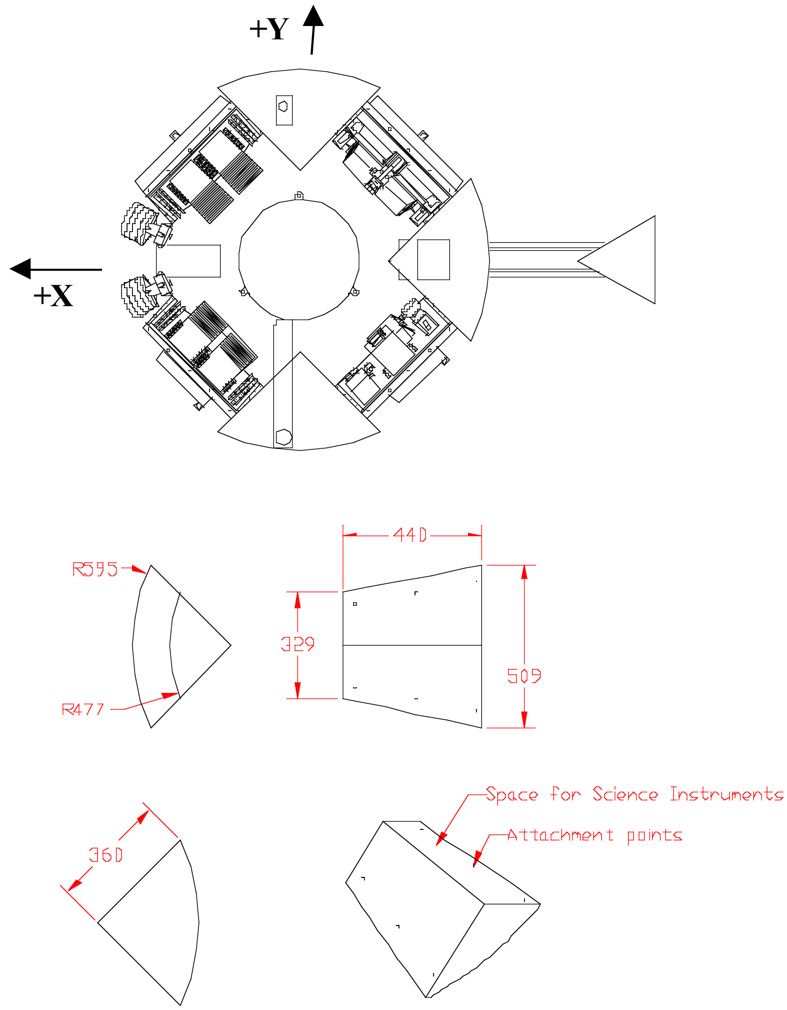

Figure 8. Bus Tapered Wedge Quadrants (dimensions in mm) (conceptual design).

margin in their instrument proposal such that the defined bus instrument volume will not be exceeded. Instruments may be located on an investigator-supplied boom extending up to one meter from a wedge as shown in Figure 8 (no hardware may extend more than 1.6 m from the spacecraft center line); any shading or thermal control required is the responsibility of the instruments.

Nadir-viewing optical instruments are to be located in the +Y and –Y tapered wedges. These two volumes are located where light baffles can be placed in the HGA/Thermal Shield System to allow for direct viewing of the Sun. The light baffles must be greater than a certain length to insure that the heat input to the instruments and bus are acceptable (see Figure 10). Also, the optical baffles must be as far as possible from the high-gain antenna beam. These constraints require that the instrument apertures be as near as possible to the outer edge of these volumes as exemplified in Figure 8.

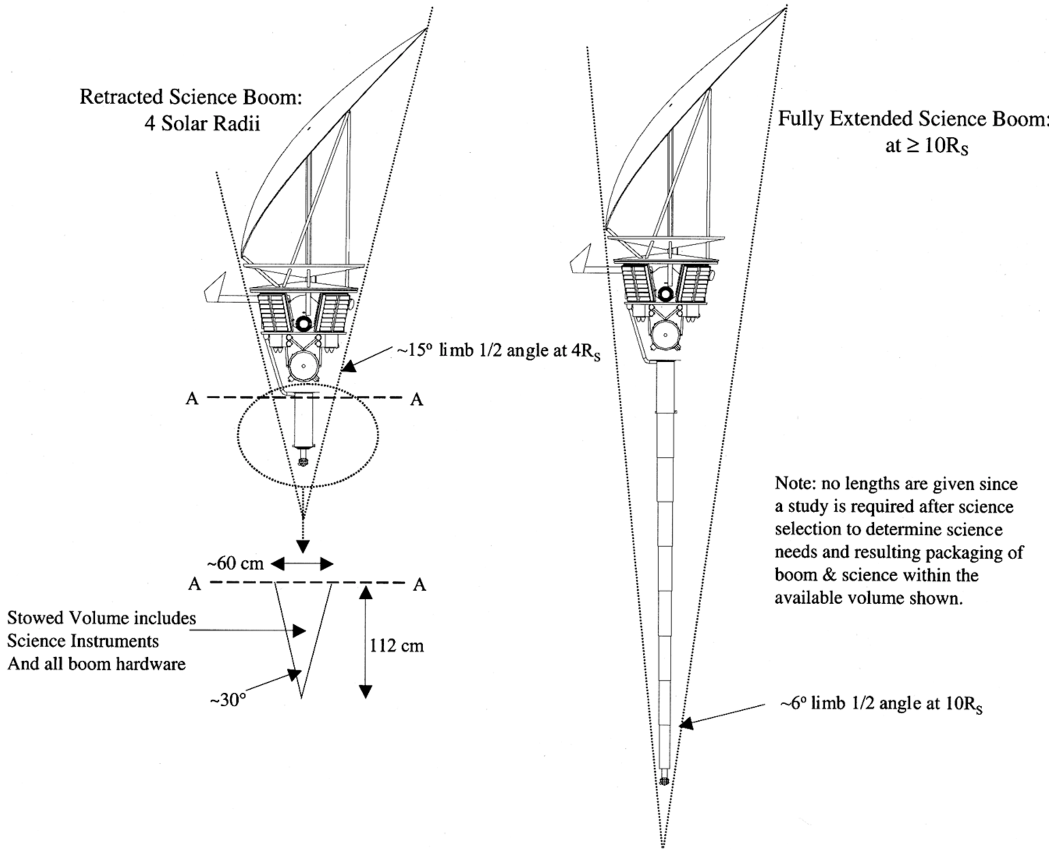

Figure 9. Integrated Science Payload boom volume. (Note: the boom approach is conceptual and could change during the conceptual design phase to best meet the science objectives within the constraints of the program.)

The volume allocated to the instruments mounted on the aft boom is a cone defined by a combination of the umbra at perihelion, the maximum extended length of the aft boom, and the attitude control angle (see Figure 9). This volume accounts for both the perihelion (4 R_S) position and the fully extended position at 8 R_S. This volume includes the instruments, the boom actuator, and the boom structure. The required volume for boom hardware is currently unknown, but for purposes of this proposal, the volume of the aft cone allocated for science instruments is as given in Table 7. This allocation assumes a strawman 4-m extendible boom. If proposed instruments do not need such large boom extensions, the base of the aft cone science volume could be increased to as much as 36 cm.

2.2.2.6 Thermal

All instrument hardware located internally to the bus shall be capable of an allowable flight operating and non-operating temperature range of -20°C to +50°C. The maximum thermal dissipation for each wedge in the bus is 28W. This maximum thermal dissipation includes all solar heat absorbed by the instrument directly or through the light baffles in the HGA or radiated or conducted from any boom extending outside of the primary sun shield/antenna umbra in addition to the electrical power thermal dissipation.

For the aft instrument boom, the maximum power dissipation is limited to the maximum heat that the instrument can radiate to space. Estimated minimum boom temperature at Jupiter is ~100 K. Minimum survival temperature for boom-mounted instruments at Jupiter must be addressed in the instrument thermal design.

All instruments are responsible for any temperature-control electrical heaters or thermal radiators located on or within the instrument package or specifically required for the conduct of the science experiment. The Project will supply only temperature sensors related to the health of the spacecraft. Any instrument need for additional heat via Radioisotope Heater Units (RHUs) should be specified in the proposal.

The –X wedge is expected to be thermally protected and isolated from the remainder of the bus to accommodate higher than usual heating due to an attached science boom and instrument that extends outside the sun-shield umbra.

In addition to electrical power, the ARPS thermal dissipation could be utilized to heat the bus, if additional heat is required. In addition to the ARPS waste heat, the spacecraft may use Radioisotope Heater Units (RHUs), electrical heaters, the thermal heat shield subsystem, louvers, radiators, and thermal blankets for temperature control. As part of the thermal integration of the instruments on the spacecraft, RHUs, if needed, may be provided near or adjacent to the instruments; however, provision for RHUs shall not be designed into any instrument.

The spacecraft thermal design will be capable of maintaining the propulsion subsystem within a 5°C and 50°C temperature range and the bus within a -20°C and 50°C temperature range throughout the mission. The current direct mission has solar distance extremes from 0.02 AU ($4R_S$) to 5 AU.

2.2.2.7 Command, Control, and Data

The spacecraft data subsystem is being developed by the X2000 program and is centered around 2 system flight computers (SFC) sharing engineering tasks and science tasks such as data processing, editing, compression, etc. The SFC will control one redundant high-speed and one redundant low-speed data bus. The protocol standard for the high-speed bus is IEEE 1394. The protocol standard for the low-speed data bus is I^2C.

A generic microcontroller will serve as the standard interface between the data buses and remote terminals such as instruments. Each microcontroller will provide interfaces to the four data buses: prime high-speed, backup high-speed, prime low-speed, backup low-speed. Two microcontrollers each will be supplied by the spacecraft for use by the remote sensing and *in situ* instrument packages (total of four). Their characteristics are defined below and in the Description Of X2000 Components Available For Use In Instrument Proposals document of the Outer Planets Program Library, available over the Internet through URL http://outerplanets.LaRC.NASA.gov/outerplanets. Software can be downloaded from the SFC into the microcontrollers for use by the instruments. The mass, power, and cost for these microcontrollers will not be charged against the payload resource allocations of Table 7 in Section 3.1. Any science data processing software that runs on the microcontrollers or the SFC must be supplied and budgeted by the science investigation, however.

The spacecraft data subsystem will include some means of bulk data storage. The current baseline design employs nonvolatile flash memory (NVM).

The planned software operating system for the spacecraft is VxWorks. The planned programming language is C^{++}. Additional middleware and other capability to access system services and to support required system interfaces will also be provided.

Tentative key requirements for the total data subsystem are:

System processor speed	>100 MIPS
High-rate bus bandwidth	100 Mb/s
Low-rate bus bandwidth	100 kb/s
Data storage	less than 6 Gbits

Only a fraction of the data subsystem capabilities defined above will be available to support science tasks as reflected in the resource allocations of Table 7 in Section 3.1. The avionics system currently baselined for these missions includes several new technology developments. The allocations listed in this document for science use are derived based on known capabilities of the fallback options that may be used in the event that the new technologies are not available within the time frame required. Thus, these allocations may not reflect the current advertised baseline capabilities. A worst-case fallback option might involve a computer with as little as 30 MIPS processing speed; in that case, multiple computers could be included to meet the science processing allocation. The data subsystem is intended to be compatible with the inclusion of additional memory and/or computing capability within a science instrument.

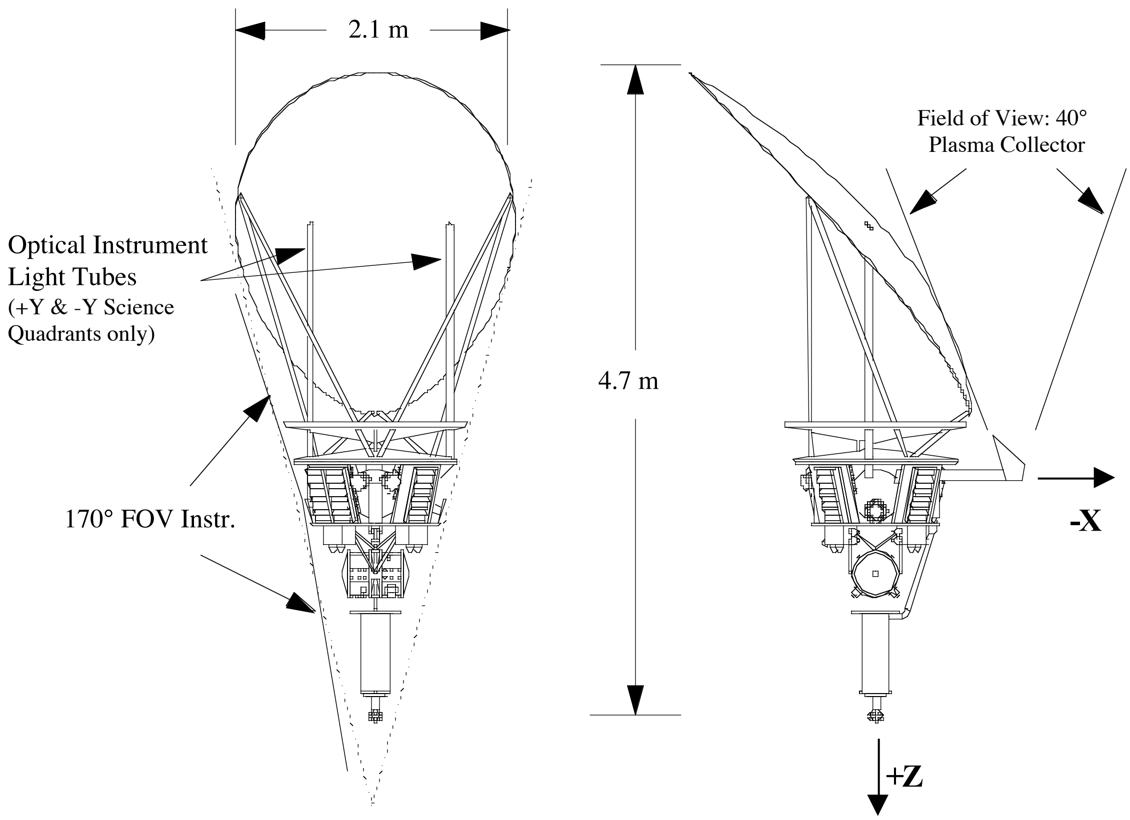

Figure 10. Bus Instrument fields of view. (Note: the boom approach is conceptual and could change during the conceptual design phase to best meet the science objectives within the constraints of the program.)

2.2.2.8 Fields of View

The field of view (FOV) for the bus-mounted instruments is 85° half angle on the tapered wedge surface as shown in Figure 10. This FOV surface is good for sensors as well as radiators. Since materials for those structures at the edge of the available stray-light FOV are still to be determined, worst-case surface thermo-optical properties should be assumed. In addition, a FOV for a nadir-viewing instrument is shown. A ±20° FOV is shown for an instrument of this type that has its own primary thermal shield mounted on a side boom. This FOV could be increased for a longer boom length. For the aft instrument boom, the maximum FOV from the tip of the boom at the fully extended position is 348°. For the fully stowed position, the FOV is 330° (Figure 9).

Given the conceptual nature of the spacecraft design, these FOVs should be considered to be the same in the Y-Z and X-Z planes. If a –X axis fixed-boom instrument is selected, the FOV from the tip of the aft boom along the –X direction could be reduced by roughly 10° in the retracted position and by roughly 4° in the extended position. Multiple extensions and retractions of the boom are expected. The boom can be positioned anywhere between fully extended and fully retracted in this conceptual design; however, the feasibility and reliability of stopping at intermediate extension positions will be re-evaluated during the spacecraft preliminary design phase. Proposers may suggest alternative aft boom designs that are compatible with spacecraft constraints; however, no mass reallocations to the instruments should be assumed as a result of alternative aft boom proposals.

If proposers deem extension of items (booms, antennae, etc. other than the –X boom described in Sec. 2.2.2.5) beyond the spacecraft umbra to be necessary to accomplish the Group 1 science objectives, they must demonstrate that this will not impact potential coronagraph viewing. They should also attempt to minimize any adverse effects on spacecraft thermal design and attitude control design recognizing that if it is later determined that such structures cannot be accommodated on the spacecraft, the proposed investigation will not pass Science Confirmation.

2.2.2.9 Coordinate System and Mechanical Design

The spacecraft coordinate system is as shown in Figure 11. The spacecraft Z axis is located through the centerline of the spacecraft with +Z in the direction the thruster nozzles are pointed (-Z is pointed at the Sun at perihelion). The X-Y plane intersects the Z axis at the base of the bus and is oriented with +X in the direction of the high-gain antenna beam boresight.

The strawman, conceptual flight system configuration, shown in Figures 2 and 6, consists of the High-Gain Antenna (HGA)/Thermal Shield, the bus, which includes engineering avionics, bus-mounted instruments, the Aft Instrument Boom, the propulsion subsystem, and the power source.

The HGA is used as the primary heat shield. Conical secondary shields are located between the primary shield and the bus. All of the shields are made of various types of Carbon-Carbon. The HGA dish and secondary thermal shields make up the HGA/Thermal Shield subsystem. The unusual dish shape of the off-axis HGA paraboloid is consistent with the quadrature geometry at the first perihelion; quadrature allows for a real-time RF link with Earth at perihelion for the first fly-by. Included in the HGA/Thermal Shield System are two light baffles. The light baffles, made of Carbon-Carbon, allow solar light to be attenuated as it passes through to the instruments that are maintained at room temperature inside the bus.

Below the shield is the "eight" sided bus as shown in Figure 8. Four sides are referred to as bus panels and house the spacecraft avionics. In between the four panels are four tapered wedges. Three of the tapered wedges are for science use as stated in Section 2.2.2.5, and the fourth wedge is used to house the attitude control sensors. The panel material (composite or metal) is to be determined. The four tapered wedge areas will interface with the panels. Instrument interface attachments will be determined after the instruments are chosen. The method of attaching the panels and wedges, along with the bus top and bottom, will require the system to be electrically sealed in order to form a Faraday cage. Loads from the HGA will

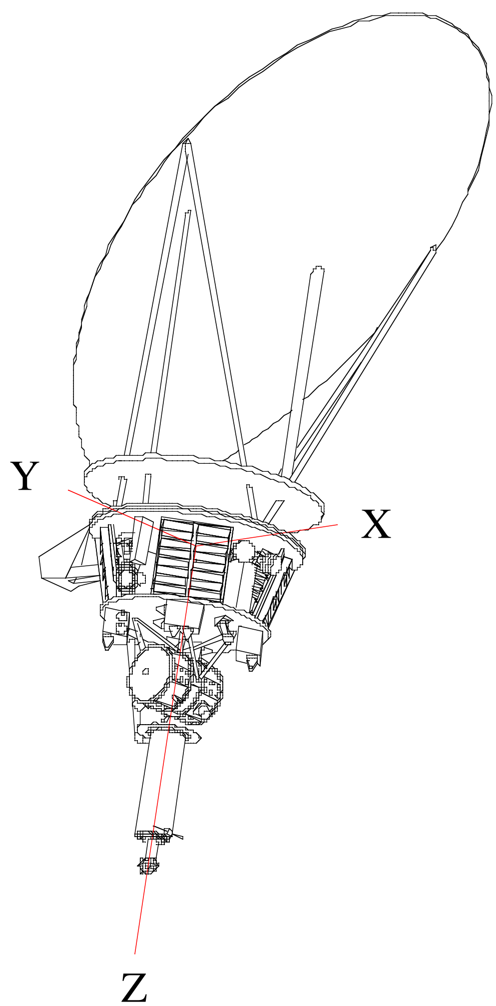

Figure 11. Solar Probe spacecraft coordinate system

be transferred through the panels and possibly the wedges depending upon the interface design.

Mounted to the base of the bus is the propulsion subsystem. The current propulsion subsystem is a single-tank, mono-propellant system utilizing hydrazine. The tank will be structurally mounted to the bus close-out plate and located inside the bus structure. The close-out plate will house all of the propulsion components including the four thruster clusters. The close-out plate will also have the integrated science payload boom attachment and the Advanced Radioisotope Power Source (ARPS) attachment bracket.

The aft instrument boom extension actuator is located on the boom within the volume described in Section 2.2.2.5. A limited number of boom extensions and retractions can be performed during the course of the mission in this conceptual design. The conceptual boom itself is assumed to have no thermal or electrical conductivity. The instruments on the boom are located very close to the ARPS, which produces radiation (gamma and neutron) and a significant magnetic field. These radiation fields are described in the Environmental Requirements document of the Outer Planets Program Library, available over the Internet through URL http://outerplanets.LaRC.NASA.gov/outerplanets. Spacecraft magnetic fields will be limited as specified in the Environmental Requirements document of the Outer Planets Program Library, available over the Internet through URL http://outerplanets.LaRC.NASA.gov/outerplanets. The instruments are also located in an area that will have some minor thruster plume impingement from the Z-axis thrusters. This impingement is not a thermal issue but a contamination issue. The disc shade below the ARPS will help reduce contamination.

Since all cabling to the aft instrument boom must cross the boom actuator, proposals should include a wire count for each instrument on this aft boom. Since larger cables will drive the sizing of the actuator, instruments are encouraged to minimize the wire count over the actuator.

The flight spacecraft will separate from the launch vehicle upper stage adapter at the base close-out plate. The current design uses pyro actuated separation nuts.

(Note: During the conceptual design phase, the instrument and spacecraft teams will revisit the design and reduce the complexity and the cost of the Solar Probe mission in order to assure that the mission will fit within the program constraints.)

2.2.2.10 Attitude Control

The Solar Probe spacecraft will be 3-axis stabilized. Attitude determination will be done using star trackers, gyros, and Sun sensors. Each of these sensors will be block redundant. Gyros will be part of an inertial reference unit. Attitude control and delta-V maneuvers will be accomplished by firing the 0.9-N thrusters of the propulsion subsystem.

Additional functions of the spacecraft attitude control subsystem are to navigate and control the Star 48V injection kick motor. Roll control during injection will be provided by the spacecraft.

Fine pointing will be accomplished using the star tracker for attitude knowledge. Nearly continuous attitude estimation is planned. The star tracker is required to provide 4π steradian attitude determination.

The gyros will be used during maneuvers and for attitude reference during the perihelion passage. The Sun sensor will be used principally for attitude acquisition during cruise and faults. Nadir pointing of the spacecraft -Z axis will be autonomously maintained.

Key baseline requirements for the overall attitude control subsystem are:

pointing accuracy (3σ)	5 mrad
pointing knowledge (3σ)	1 mrad (absolute in inertial hold)
	3 mrad (absolute while slewing)
pointing stability (3σ)	1 mrad in 1 sec
	20 μrad in 10 msec

2.2.2.11 Telecommunications

The telecommunications subsystem for Solar Probe reference mission consists of a parabolic high-gain antenna, block redundant 3-watt RF X-band Solid State Power Amplifiers (SSPAs), and block redundant Space Transponding Modems (STMs). A top-level diagram showing the telecom subsystem architecture is shown in Figure 12. The telecommunications configuration shown is a unified uplink/downlink X-band design such that all telecom link functions can be utilized simultaneously.

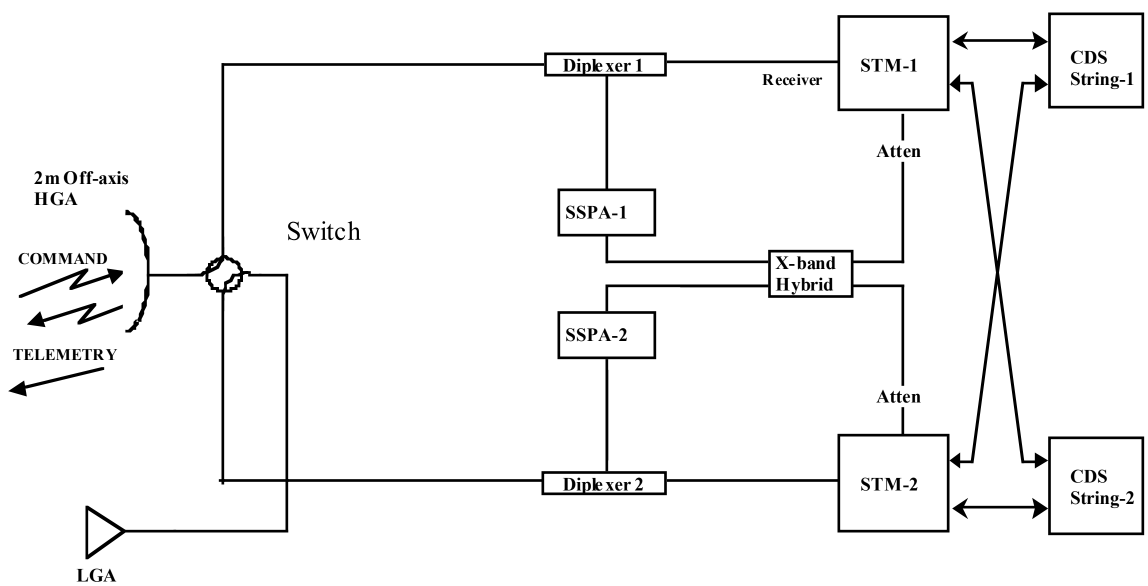

Figure 12. Telecomm subsystem architecture

Since both the DSN and flight system have constant power transmitters, the division of power between simultaneous links will vary depending on specific link configurations. This will affect link performance when supporting multiple links at once. Key communications parameters for the Solar Probe mission at perihelion are listed in Table 5.

Table 5. Solar Probe telecommunications parameters at perihelion

Parameter	*Solar Probe*	*Units*
Transmitter Power	3	Watts
High Gain Antenna	40	dBi-RCP
Low Gain Antenna	6	dBi-RCP
Science Uplink Command Rate	20	bps
Typical DSN Lockup Time	5	min
Downlink rate (maximum)	50	kbps

Downlink rate is at perihelion and assumes 70-m DSN antenna at 20º elevation angle and 90% weather. Uplink command rate assumes 70-m DSN transmitting at 20 kW to the HGA and represents the effective transmission rate for science commands (the actual bit rate sent to the spacecraft is substantially higher); however, uplink commanding is not possible within ±6 hours of perihelion due to solar interference.

A representative real time telemetry rate near perihelion is shown in Figure 13. These data are consistent with the assumptions shown in the figure. An additional fundamental assumption is that the amplitude scintillations caused by coronal perturbations on the downlink are infrequent transient events and will not affect this average telemetry rate performance. Note that the effective downlink rate allocated for science data return in Table 7 is less due to overhead (packetizing, coding), solar scintillation effects, engineering telemetry, and reserve.

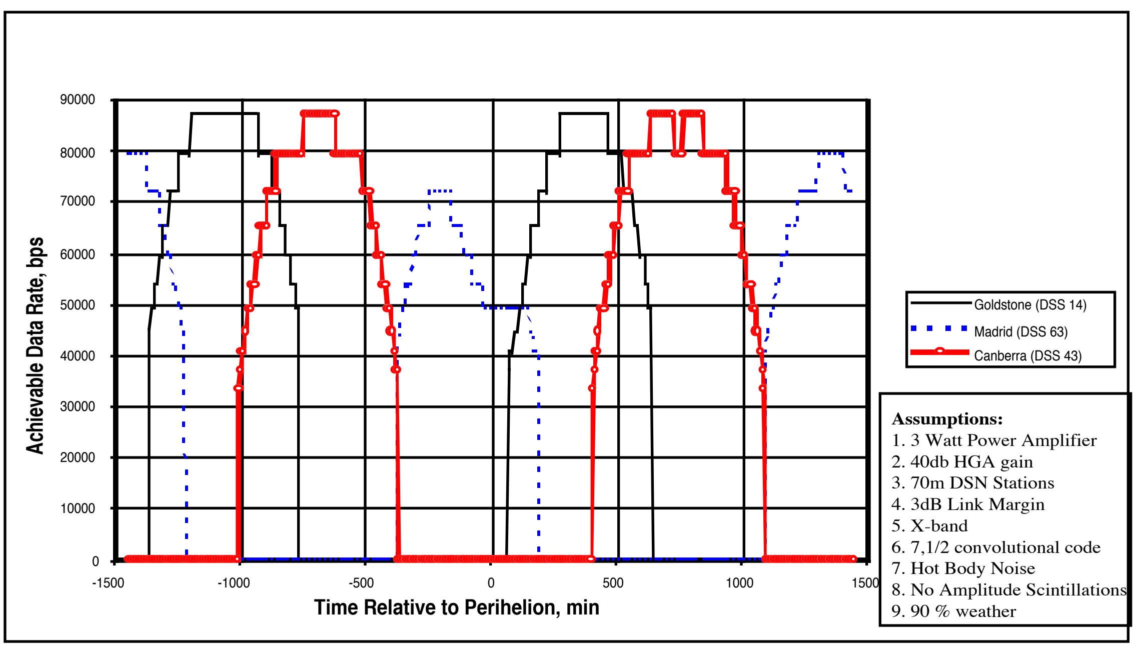

Figure 13. Solar Probe telemetry rate near perihelion for quadrature geometry

2.2.2.12 Propulsion

The propulsion subsystem will provide the required onboard incremental changes in velocity and reaction attitude control capability for the spacecraft over the lifetime of the mission. The total propulsion delta-V is baselined at 90 m/s. This is sized for the Jupiter gravity-assist trajectory reference mission with two 4-solar-radii flybys of the Sun. A mono-propellant hydrazine system is utilized. 0.9-N thrusters are utilized for both propulsion and attitude control.

2.2.3 Launch Vehicle

2.2.3.1 Launch Site

The expected launch site will be either the NASA Kennedy Space Center or the U.S. Air Force Cape Canaveral Station, Florida, USA.

2.2.3.2 Launch Vehicle

The final launch vehicle selection has not yet been made. The reference mission assumes that the baseline launch vehicle is a Delta 3/Atlas 3-class with a Star 48V upper stage. It is possible that the launch system will be changed to one of the Delta-IV/Atlas V-class plus Star48V upper stage; any such change, and updates to launch environments and other relevant parameters, will be posted in accordance with Sec. 2.11 of the main body of this AO.

2.2.4 In-Flight And Near-Sun Environmental Hazards

Generally recognized environmental hazards for Solar Probe fall into three categories:

1. Radiation environment;
2. Dust impacts; and
3. Sublimation from the carbon-carbon thermal shield/antenna.

Hazards (1) and (2) are functions of the ambient (natural) environment; hazard (3) is self-induced by the spacecraft's presence. The level of all three hazards as well as necessary mitigation levels and procedures have been the subject of ongoing debate since the Solar Probe mission was first proposed in the late 1970s. The earliest work was done in conjunction with the incarnation of Solar Probe as the Starprobe mission in the early years (Neugebauer et al., 1978, on the topic of radiation; Goldstein et al., 1980, on the topics of outgassing and spacecraft potential); the most recent comprehensive work was completed at the Solar Probe Environment Workshop [Proceedings of the First U.S.-Russian Scientific Workshop, ed. O. Vaisberg and B. Tsurutani, 1995] held in Moscow, Russia.

These three environmental hazards may further be grouped in order of increasing problem levels:

1. Measurement Contamination - including obscuration of optics and detection of spacecraft-generated signatures in the *in situ* measurements;
2. Measurement Obscuration - measurements dominated by the hazard environment, including both spacecraft-generated signatures and processing and detection failures in electronics caused by an increased radiation background
3. Instrument Failure - e.g. arcing, structural damage from grain impacts, permanent electronics failure from radiation damage; and
4. Spacecraft Failure - Structural failure and/or avionics failure producing the loss of the spacecraft and the mission.

The maximum acceptable hazard level is just prior to encountering level (2), i.e., contamination of measurements is taken as acceptable - but this implies that the contamination can be recognized and worked around or calibrated out.

Other environmental requirements are defined in the Environmental Requirements document of the Outer Planets Program Library, available over the Internet through URL http://outerplanets.LaRC.NASA.gov/outerplanets.

2.2.4.1 Dust Hazards

There are no data on the magnitude of the dust environment near to the Sun. Observations of scattered light (the F-corona) suggest the presence of dust near the Sun but yield no information on the size distribution, and there is ambiguity in separating thermal from scattering effects in the measured light intensities (Mann and MacQueen, 1995). Within 0.3 AU of the Sun, heating and sublimation of dust is expected to lead to a depletion in the dust environment (dust originates from a variety of sources and is decelerated on Keplerian orbits by the Poynting-Robertson effect) (Mann, 1995). Observations of zodiacal light from the Helios spacecraft suggest that a conservative extrapolation can be made using $r^{-1.3}$ with most of the dust concentrated toward the plane of the ecliptic with an exponential scale-height distribution (Tsurutani and Randolph, 1990; Skalsky and Andreev, 1995). Extrapolations based upon this model suggest a worst case mass flux in ~micron size particles of 10^{-9} g m^{-2} s^{-1} at 4 R_S. A random hit at typical expected speeds of ~200 km s^{-1} could, of course, cause structural failure of the spacecraft.

Figure 14 shows the current best estimate of the integral dust particle fluence on the Solar Probe spacecraft over the entire mission.

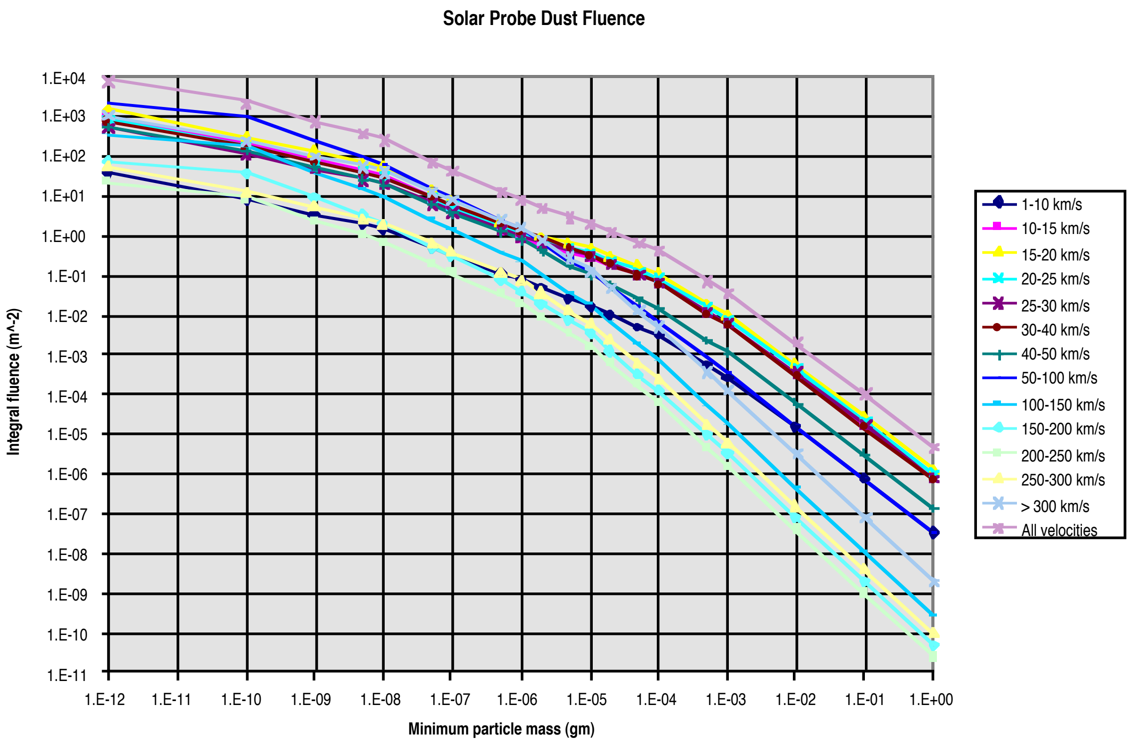

Figure 14. Integral Solar Probe dust particle fluence

Table 6 gives the current best estimate of the expected fluence of particles with masses and velocities great enough to penetrate 100 mils of aluminum assuming a particle density of 2.5 gm/cm^3. Fluences are shown for surfaces having random orientation in space, oriented normal to the spacecraft velocity vector (+v), and oriented normal to the spacecraft negative velocity vector (-v). The great majority of the potentially damaging particle impacts will occur during the perihelion passages (inside 0.36 AU) on surfaces facing the spacecraft velocity direction, which will be on the +Y side of the spacecraft, parallel with its Z axis. After science selection, the PIs and the spacecraft team will work together to determine how to protect the instruments against such micro-meteoroid impacts.

Table 6. Fluence ($number/m^2$) of $2.5\text{-}gm/cm^3$ particles on the Solar Probe spacecraft that will penetrate 100 mils of aluminum

	Surface orientation		
Time period	random	+v	-v
Through first perihelion	22.8	75.4	0.0023
Through second perihelion	46.4	153.4	0.0071

2.2.4.2 Radiation Hazard

The Solar Probe mission is subject to three different radiation hazards: the magnetospheric environment of Jupiter, the interplanetary environment, and the solar coronal environment itself. A Jupiter flyby is required for any Solar Probe mission using near-term propulsion technology due to the requirement for eliminating the angular momentum of the probe acquired by the launch from Earth in order to approach closely to the Sun. The secondary need for a Jupiter flyby is the rotation of the heliocentric probe orbital plane to ~90 deg to the plane of the ecliptic. Launch dates over the next two decades require a Jupiter flyby to within 10 Jovian radii (and as close as ~5 R_J). This region has been well explored by the Pioneer 10 and 11, Voyager 1 and 2, Galileo, and Ulysses spacecraft.

For a ~10 R_J flyby distance, the expected radiation environment is ~35 krad with 100 mil of Al shielding. By using approved parts lists and introducing functional redundancy of appropriate subsystems, this level of radiation background is easily dealt with. Similar remarks apply to single event effects produced by galactic cosmic rays and/or solar proton events (Garrett, 1995).

The radiation hazard provided by the Sun itself remains unknown. Both shock acceleration and direct (flare) acceleration have been implicated in producing particles seen in the 10-100 MeV energy range. Work by Tsurutani and Lin (1985) and Reames (1995) has indicated that the dominant component of the proton flux at 1 AU is due to shock acceleration. Such shocks occur ahead of fast Coronal Mass Ejections (CMEs), which occur primarily during solar maximum. Ulysses has indicated that CME-driven shocks can exist at high heliolatitudes (Gosling et al. 1998) with significant particle acceleration occurring.

Wu, et al. (1995) have indicated that CME-related shocks first form at a substantial distance from the Sun (typically 15-20 R_S). Additionally, since high Mach number shocks are more effective at accelerating energetic particles, the near-solar particle fluences would be less than an inward scaling of r^{-2} (fluence) or r^{-3} flux (Feynman et al., 1995).

Solar Probe will be approaching the Sun from high latitudes and will pass over the near-equatorial active regions relatively quickly at perihelion. Kiplinger and Tsurutani (1995) have examined the probability of a flare occurring when Solar Probe is within ±30 degrees of the solar equator. Using the statistics of Reames [1995], they find the probability to be much less than 1% during solar minimum and about 2% during solar maximum.

Clearly the particle fluence/flux due to CME shocks at high latitudes during solar maximum also needs to be studied more closely to better understand the probability of occurrence during Solar Probe passage and the particle quantitative doses.

2.2.4.3 Outgassing-Sublimation Hazard

Outgassing and sublimation can pose hazards to the Solar Probe in several ways. The most important likely problem is that neutrals released from the high-temperature heat shield will become ionized close enough to the spacecraft to either alter the properties of the solar wind ions and electrons or to generate plasma waves that might mask observation of ambient plasma waves. An additional issue is contamination of spacecraft surfaces by deposition of neutral carbon. If the density of the neutral carbon gas is sufficiently low that the flow of carbon neutrals is collisionless, this would not seem to be a major problem because sensitive surfaces could be protected by restricting the line-of-sight to the hot neutral source. Proper prelaunch heat treatment can reduce the risk from outgassing, but as an unavoidable minimum one is left with sublimation. Sublimation rates are discussed in the subsection 2.2.4.4, and in 2.2.4.4.1; the ion pickup process and estimated mass loss rates are discussed and found to be sufficiently low to prevent interference with the science observations.

2.2.4.4 Sublimation Rates

In the design of the thermal shield, the logic has been the following: (1) sublimation of shield material (carbon) could interfere with measurements of the *in situ* environment (such measurements are the rationale for the mission); (2) shield sublimation is a function of shield temperature; (3) shield temperature must be driven by the "acceptable" outgassing/ sublimation/ablation rate - as determined by another calculation; (4) shield temperature is then determined by the amount of solar loading versus the amount of radiative area. The actual calculations of shield temperature include both radiation and conduction (which is much less important). For the planned Solar Probe heat shield, which also functions as an antenna, there is a hot region near the tip of the shield that dominates sublimation. Measured sublimation rates from graphite have been available for some time [Drowart et al., 1959]. There are preliminary indications (Valentine et al., 1997) that outgassing from various carbon-carbon matrices is about an order of magnitude less, presumably due to surface energy effects.

The tip of the spacecraft heat shield is approximated as having 0.4 m^2 at 2242 K and 0.6 m^2 at 2204 K, which provide the major portion of the strongly temperature dependent sublimation. For SAIC and C CAT carbon-carbon material (low sublimation rate), the loss rates are 0.0046 mg/s m^2 at 2242 K, and 0.0015 mg/s m^2 at 2204 K. Since mass spectrometry data was not available in the Valentine et al. (1997) study, the Joint Army-Navy-NASA-Air Force (JANNAF) thermochemical tables are used to estimate the relative amounts of loss of various multiatomic carbon neutrals (C_1, C_2, C_3, C_4, and C_5). The total mass loss rate for the current Solar Probe design is estimated to be about 3.3 10^{-3} mg/s, if the Valentine et al. (1997) study results are used; if the JANNAF tables are used the results would be about 5 times greater (1.6 $\times 10^{-2}$ mg/s). We note that this is the maximum sublimation rate that occurs at 4 R_S. When the spacecraft is farther from the Sun, the temperature decrease leads to orders of magnitude less sublimation.

2.2.4.4.1 Mass Loss Rate and Interference with Science Objectives

Early on in Solar probe concept studies, it was recognized that the composition of the thermal shield would have a driving effect on how close to the Sun that the Solar Probe could approach before the *in situ* measurements would be corrupted. Goldstein *et al.* [1980] noted that the driving criterion was "...a requirement of no important interference with scientific objectives." In particular, they were concerned that the impact on plasma wave and electron observations be minimal. Positive ions can presumably be separated from *in situ* ions in the plasma measurements on the basis of ionization state and composition. However, sufficiently large mass loss rates could alter the local electric field in the vicinity of the spacecraft adversely impacting plasma and especially electron measurements. Based upon the criterion that the spacecraft float to no more than 20 V with respect to the local plasma (and introducing a safety factor of 5), they derived a maximum acceptable outgassing rate of 3.0 mg s^{-1}. An independent constraint based upon less than a 1% chance of an electron collision with outgassing carbon was an order of magnitude less stringent. Plasma wave and wake effects were not found to be important at this outgassing/ sublimation level. We note that the recent carbon-carbon material test indicates sublimation rates far below this value.

The question of pickup ion effects was investigated by Okada et al. (1995), Goldstein (1995), and Tsurutani et al. (1995). Goldstein looked at the possibility that the pickup plasma would interact with the solar wind plasma via waves that stand in the spacecraft frame. On this basis, the waves of interest are lower hybrid waves and electron cyclotron waves. From the wave impedance for these types of waves, Goldstein (1995) estimated that a mass loss rate of 2.1×10^{-2} gm/s would result in a maximum potential perturbation in the plasma of about 5 volts, but because of uncertainties in the method of calculation, recommended that the mass loss rate be limited to about 2×10^{-3} gm/s. This work assumed a neutral ionization time of 30 seconds and that the mass was dominated by C_3 ions. Okada et al. (1995) examined the possibility that C_2^+ ions and related electrons might generate plasma instabilities. Using the Kyoto University Electromagnetic Particle Code (KEMPO), they found that there were no substantial waves generated by either the ion or electron pickup. The combined U.S.-Russian panel on Atmospheric and Electromagnetic Environment Group (Tsurutani, et al., 1995) has determined that the carbon/electron pickup process seems to not be a problem for Solar Probe.

The Science Definition Team obtained some simple checks on the above work.

As a check on the ionization rate assumed in the previous studies, W.-H. Ip independently calculated the ionization rates using more recent estimates of photoionization rates and electron impact rates. For photo-ionization and electron impact rates near the Sun he obtained:

Photo-ionization and electron impact rates at 4 solar radii

	Photo-ionization	Electron impact	Total (a)	Total (b)
C_1	2.4×10^{-2} /s	1.34×10^{-3} /s	2.53×10^{-2} /s	3.07×10^{-2} /s
C_2	2.6×10^{-3}	2.38×10^{-3} /s	4.98×10^{-3} /s	1.45×10^{-2}
C_3	2.6×10^{-3}	2.38×10^{-3} /s	4.98×10^{-3} /s	1.45×10^{-2}

Note: The C_1 and C_2 photo-ionization rates are from W. F. Huebner et al., (1992). The solar condition was assumed to be for the quiet Sun at solar minimum. There exist no laboratory data for the photo-ionization cross sections of C_2 and C_3. The electron impact ionization rates for C_1 and C_2 were obtained from D. Shemansky (private. comm., 1997). As with photo-ionization, the electron impact rate of C_3 is assumed to be the same as that of C_2. The electron temperature is assumed to be 10^6 K and case (a) is for electron number density of 10^4 /cm^3, and case (b) is for 5×10^4 /cm^3.

The effect of mass loading upon directly decelerating the solar wind was found to be negligible. Within about 2 meters of the spacecraft, the pickup ion number densities were found to be comparable to the solar wind proton densities, but this would not affect the observations. (Note this conclusion is based upon the old, higher outgassing estimates based on JANNAF tables rather than the lower estimates obtained in the Valentine (1997) report.). In view of these results, the most likely (if any) source of interference with the measurements would be generation of plasma waves by the pickup ions, thus confusing the interpretation of the waves normally present in the solar wind. It was assumed that the lower hybrid (modified two-stream) instability would be the most likely source of wave growth. This instability typically requires pickup ion density to be about 10% of the ambient ion density (at least, if the instability is to be isolated in the frequency spectrum). For the case of encounter at four solar radii, the maximum growth rate was taken as 0.5 w_{LH} (w_{LH} = lower-hybrid frequency), and the minimum growth length was taken as the solar wind velocity divided by this growth rate. On this basis, the minimum growth rate was found to be 120 m, and full growth to saturation is typically found only after 30/ w_{LH}. As the scale size of the ion cloud where the density is 10% or greater is much smaller than 120 m, it is concluded that the lower hybrid instability is not likely to be a cause of interference.

2.3 Mission Development Concept

2.3.1 Flight System Design and Deliveries

Though the three OP/SP spacecraft will be launched over a period of 3-4 years, the initial spacecraft design will be performed by the same personnel assigned to a joint design team. This team will continue into the detailed design of the Europa Orbiter and Pluto-Kuiper Express spacecraft while identifying areas of commonality for incorporation later into the detailed design of the Solar Probe spacecraft. Common subsystem designs will be used wherever possible to minimize the cost of developing and testing each spacecraft.

The OP/SP Project expects to employ the JPL Mission Data System (MDS) as its end-to-end data system. The MDS is currently under development and comprises both flight and ground software used by multimission and project personnel to operate the spacecraft. MDS will be used in software development, system test, and in actual mission operations and will enable the missions to collect, transport, store, and act on both commands and telemetry. The MDS software architecture employs an object-oriented approach. The MDS spacecraft component will provide a standard interface to the science instruments including time synchronization, commands, data acquisition and storage, system coordination, fault protection, memory loading, and diagnostic functions. The software architecture is designed such that a core set of software functions are coded and used for all missions. Some mission-specific software will be required to specifically address those unique aspects of each mission, spacecraft, and payload. This core architecture will allow for software reuse, reduced cost in the development and testing of the software, smaller flight operations, faster sequence turn-around times, and improved science return in the event of required failure recovery responses.

Science proposers who intend to exploit available spacecraft computer resources will need to be compatible with the MDS software architecture and design, at least for software that is resident in the Spacecraft Flight Computer (SFC) and Generic Microcontroller. The extent to which any instrument flight software that runs on an internal instrument computer or any investigator-generated ground sequence planning, Ground Support Equipment, or data analysis software will need to adhere to MDS standards will be specified in an OP/SP Software Management Plan. Instrument proposers should plan to have at least one software expert in residence at JPL for at least 6 months prior to instrument PDR for training in the MDS methodology, development environment, and tools. MDS coding will be in C^{++}, and the operating system is VxWorks/Tornado. For the purposes of this AO, it may be assumed that the required software licenses will be provided by the Project. MDS documentation will be provided including a Development Plan specifying the software development process, coding standards, review criteria, and configuration management approach; a Capabilities Catalog describing the capabilities supported by the MDS architecture; and a Users Reference Guide. Science instrument providers will be expected to participate in developing command and telemetry dictionaries, associated system design constraints, associated command elaboration products, and instrument flight rules and constraints.

The planned X2000 First Delivery includes multimission avionics, software, and other equipment for the three missions. The recurring cost for the flight equipment is expected to be comparatively low. The propulsion modules and science packages are unique, however, and they will be a significant factor in the total cost of those missions. These mission-unique costs are borne by each individual mission, but by using common flight support and test equipment and common ground and flight software modules, each mission can reduce its integration and test costs.

The Project will supply to instrument PIs prototype and engineering model microcontroller slices (GMCs, identical to microcontroller slices [MCS] referred to in the spacecraft functional block diagram, Figure 7) for use in simulating the spacecraft interface during their instrument development effort. PIs will need to procure hardware (per Project specifications) that will include a computer workstation (e.g., mid-range Sun), a COTS single board computer (currently assumed to be Power PC based) with an Ethernet interface, and commercial 1394 and I^2C buses to model the spacecraft functions. This hardware, in conjunction with the GMC, will host the MDS flight and ground software system with which the instrument software will need to interface. The Project will supply the MDS software system that is hosted on this hardware. A partial delivery of the Project-furnished MDS software, including the GMC operating system and device drivers, the capability to download code into the GMC, and 1394 and I^2C bus interface code, will be made available by TBD. A more complete version of the MDS software will become available on TBD.

Whenever possible, leveraging of technology developments supported by other NASA missions and/or technology development programs will be used where the capabilities match the needs of OP/SP. Such arrangements include incorporation of technologies supported by the New Millennium and Mars Programs. Some mission-unique technology (e.g., heat shield/antenna for Solar Probe) requires that OP/SP wholly support the development.

Standard, reasonable services will be provided the instruments during integration and testing at the system integrator’s facility and the launch site. These include:

- Sterile dry N_2 purge (to be connected after receipt at the system integrator). It is the Instrument's responsibility to provide this during shipment and delivery into the integrator's facility;
- Office space with telephones and modem connections; and
- Laboratory space with limited tool capability in the integration facility.

A Spacecraft Test Laboratory will be developed at the system integrator’s facility to simulate the spacecraft and software. The instruments shall provide software simulators of sufficient fidelity as well as breadboards and instrument simulators to support this effort.

2.4 Mission Operations Concept

2.4.1 Integrated Mission Flight Operations Team

The Europa Orbiter, Pluto-Kuiper Express, and Solar Probe missions will share a single core flight team and a common mission data system. This approach is enabled by the common X2000 avionics design shared by all three spacecraft together with a large percentage of common flight software. Each mission will supplement the shared operations capability with a few mission-dedicated personnel including mission planners, instrument representatives, and science investigation teams.

The current plan is for the core flight operations team to be supported by a university-based operations team, which will be competitively chosen in 2001. The university team will be delegated selected routine flight operations tasks to enhance the ability to operate multiple spacecraft simultaneously, to support educational outreach, and to provide a potential source of trained new-hires during the 15 years of flight operations. A workstation-based ground data system design makes implementation of a replica Project Operations Center (POC) at a university cost effective. Science workstations that allow science team members to interact with the operations system from remote sites will be developed as part of the ground data system design.

2.4.2 Beacon Mode Cruise

Routine Deep Space Network (DSN) tracking during cruise will be limited to a single, 4-hour pass every two weeks. This limit on telemetry and radiometric data collection and spacecraft commanding during cruise is intended to keep operations team costs low and reflects the new NASA full-cost-accounting policy, whereby missions are charged for DSN tracking time. To prevent a spacecraft anomaly from going undetected by the ground for a period of up to two weeks, a daily spacecraft beacon monitor track will be performed to establish that the spacecraft is on Earth-point and that no onboard event has been detected that requires ground interaction until the next regularly scheduled telemetry pass. The beacon signal generated by the spacecraft is a subcarrier tone that can be received by a small (5 or 10 meter) ground antenna and detected by a low-cost receiver / detector. The daily beacon monitor check for each spacecraft may be a task delegated to the university operations team.

On-board software that supports Beacon Mode operations includes fault detection and containment software that allows the spacecraft to safe itself during cruise for up to 2 weeks without ground action. Advanced engineering data summarization, onboard alarm limit checking, onboard performance trending, and adaptive anomaly data capture capabilities will also be provided.

The assumption is that science instruments are powered off during cruise except as required for instrument survival. Approximately once a year, or as negotiated with the Principal Investigators, the instruments will be turned on, calibrated, and tested, along with encounter sequence macros that have been developed during the year. Extra DSN tracking during this week will be provided to support the additional commanding and telemetry data collection required.

OP/SP data management and data transport protocols will be X2000 MDS-based and will exploit multimission TMOD data services that will have been upgraded to support the MDS design. The MDS design assumes a common flight/ground file-based data management framework. Files will be used to package and store logical data units (objects) that may not map well into the packet model. The goal is to have management of both onboard data files and ground data files appear similar to the user. File management will support long-popular storage/access capabilities for numerous types of nontelemetry data products. File-based transport protocols will be provided for both S/C-to-ground and ground-to-ground nodes. Packetization will be provided as the underlying mechanism of flight-to-ground file data

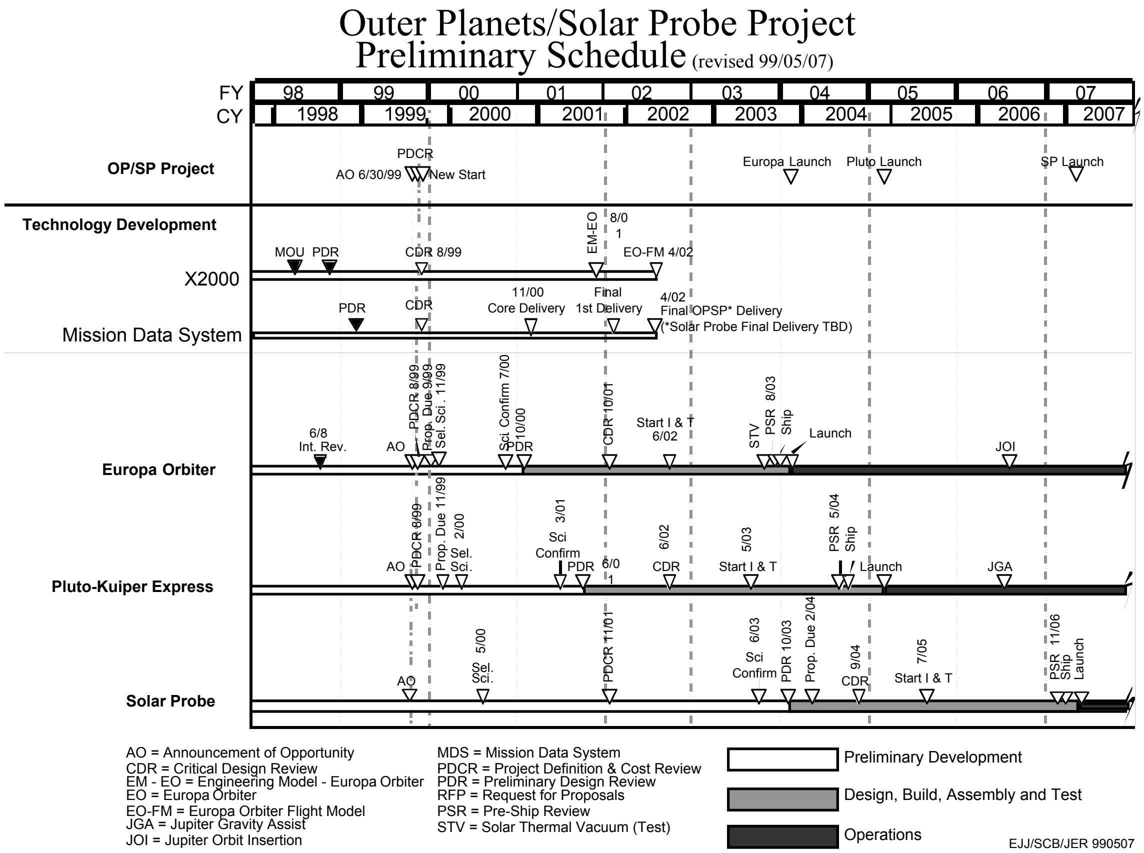

Figure 15. Outer Planets/Solar Probe preliminary schedule

transport. The goal is to make packetization invisible to file-based data management and transport. An implication of this approach is that needed time tags and other ancillary data provided in packet headers and ancillary data packets in the traditional packet-based, data-stream-based systems will have to be provided within the data objects/data files.

2.4.3 Encounter Operations

Transition from cruise operations to encounter operations for the Solar Probe mission starts at perihelion - 1 month. Starting at this time, DSN coverage will increase, along with operations team staffing to support higher activity levels and mission critical events. If available within mission constraints, operations resources will be available to support instrument calibration and serendipitous science observations during the Jupiter gravity assist flyby for the Solar Probe mission.

2.5 Project Schedule

The Project Schedule is given in Figure 15.

3. Science Investigations

3.1 Resources for the Science Investigations

Table 7 summarizes the key resource allocations for the Solar Probe science payload. Proposals that fall outside the allocations will have a lower probability of selection.

Table 7. Solar Probe science instrument key resource allocations

		Allocations	
Resource	Units	*In situ* Package	Remote Sensing
Volume	cm	36x44 wedge w/ ≤1-m side boom; 15x74 aft cone	2 36x44 wedges
Cost	M$ (real yr)	17	18
Power (average)	watts	10	5
Mass	kg	12	9
Data storage	Gbits	1.0	1.2
Computer processing	MIPS	14	16
Data rate	kbps	11	14
Bus bandwidth (asynchronous)	Mbps	12	14
Thermal Power Dissipation	watts	28/wedge	28/wedge

Any instrument covers and mounting booms other than the aft boom structure and actuator must have their mass included as part of the instrument mass even if the covers are jettisoned. The mass of any cabling from boom-mounted instruments to the spacecraft bus is considered part of the instrument mass. If an extendible aft boom is not needed to support the proposed *in situ* instrument package, the *in situ* science mass allocation is reduced from 12 to 10 kg to account for less cabling. Instruments must also provide their own thermal control system and shades, if located outside the main sunshade umbra. Any instrument purge equipment beyond fittings and internal plumbing that are part of the instrument will not have its mass charged against the above instrument allocations.

Investigations may exceed the allocated levels of data storage and computer processing MIPS by including the required extra memory or computer as part of their own hardware deliverable. X2000 parts are available for use by science investigators for this purpose, as listed in the Description Of X2000 Components Available For Use In Instrument Proposals document of the Outer Planets Program Library, available over the Internet through URL http://outerplanets.LaRC.NASA.gov/outerplanets. The cost and mass to cover use of such parts must be included in the instrument totals.

The computer processing allocation in Table 7 is for science use of the SFC. Each dedicated instrument-interface microcontroller (GMC) could potentially provide additional instrument computing capability subject to power availability constraints. Proposers, however, should not assume that this potential additional computing capability is available in developing their proposals.

It is anticipated that the teams of *in situ* and remote sensing investigators selected via this AO will be kept small for reasons of efficiency and economy. The total funding guideline in real year dollars to support these investigator teams (over and above the instrument development cost guideline in Table 7) is as follows:

Team	Development phase	Operations phase
Remote sensing	\$1.4M	\$14.6M
In situ science	\$1.4M	\$14.6M

Table 8 gives the funding profile guideline by fiscal year for each investigation (hardware plus science investigators). Note that these guideline budgets are beyond the current budget-planning horizon, which extends only to 2004, and therefore should be viewed as guidelines subject to future change and not as commitments. However, for purposes of proposals responding to this AO, these guidelines shall apply.

Table 8. Investigation (instrument and investigators) New Obligation Authority (NOA) funding profile guideline in millions of real year dollars for the development and operations phases

Instruments Development NOA Guideline											
	FY	00	01	02	03	04	05	06	07	Sum	
Remote Sensing		0.1	0.1	0.1	2.4	6.8	6.7	1.5	0.3	18.0	
In situ Science		0.1	0.1	0.1	2.4	6.3	6.1	1.6	0.3	17.0	
Science Team NOA Guideline Development Phase											
	FY	00	01	02	03	04	05	06	07	Sum	
Remote Sensing Team		0.1	0.1	0.1	0.1	0.3	0.3	0.3	0.1	1.4	
In situ Science Team		0.1	0.1	0.1	0.1	0.3	0.3	0.3	0.1	1.4	
Science Team NOA Guideline Operations Phase											
	FY	07	08	09	10	11	12	13	14	15	16	Sum
Remote Sensing Team		0.6	0.8	1.0	1.9	1.9	1.7	1.7	1.6	1.8	1.6	14.6
In situ Science Team		0.6	0.8	1.0	1.9	1.9	1.7	1.7	1.6	1.8	1.6	14.6

The list below summarizes the policies on mass, power, and cost accounting that should be assumed by proposers:

To be charged to science investigations:

- Power converters;
- Electrical thermal control heaters;
- Inflight purge equipment internal to an instrument;
- Instrument covers;
- Instrument radiation shielding;
- Science electronics cards/slices;
- Science electronics housing;
- Instrument sunshades and mounting booms (other than the aft boom); and
- Aft-boom instrument cabling.

To be charged to the spacecraft:

- Instrument interface microcontroller;
- Inflight purge equipment external to an instrument;
- All RHUs (none permitted internal to instruments); and
- Aft science boom.

3.2 Interaction with the Project

3.2.1 Project Fiscal Policy

The sections below include items that are pertinent for consideration by proposers in preparation of responses to this AO.

3.2.1.1 Budgetary Authority

NASA will annually allocate New Obligation Authority (NOA) to JPL for the Outer Planets/Solar Probe Project based on an Implementation Plan and updates submitted by the Project. In turn, the Project Office will allocate NOA annually to the Project Work Breakdown Structure primary elements based on the NASA NOA, the plans submitted by the leaders of each element (two of whom are the Chief Scientist and the Flight Instrument Development Manager), and the needs of the Project. Each mission (Europa, Pluto, and Solar Probe) has a Project Scientist, and one of them has additional duty as Chief Scientist. The Science Investigation Principal Investigators whom NASA selects through this AO will negotiate their Statements of Work (SOWs), budget submissions, and authority with the Flight Instruments Development Manager, who will be assisted in these negotiations by the appropriate Project Scientist. The resulting SOW and funding schedule will be documented in a contract between JPL and the PI's institution; this contract will be modified, if necessary, through the course of mission development and operations, covering the period of time from contract award to final delivery of science products after the end of the mission.

3.2.1.2 Mission Budget Environment

Total project costs will be a primary consideration in all design and development decisions and activities. Other requirements will have flexibility and will be prioritized to provide adequate margins and options for staying within cost and schedule constraints.

3.2.2 Project Organization

Overall project leadership and coordination is provided by the Project Manager and Project Office staff. The project is organized as shown in Figure 16. The Project Scientist is a member of the Project Office staff.

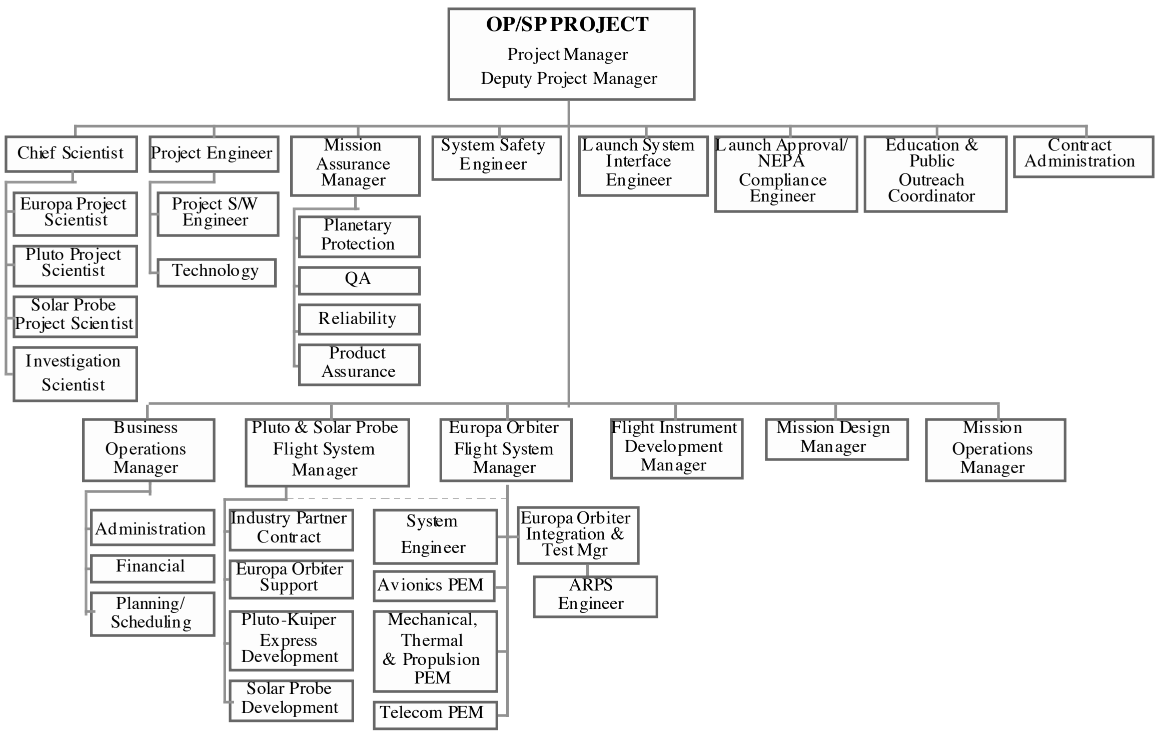

Figure 16. Organization chart for the Outer Planets/Solar Probe Project.

3.2.2.1 Science Investigators as Members of Project Teams

PIs and their lead instrument developers will become members of an integrated implementation team for their respective mission.

Primary interfaces with each mission implementation team will be in the following areas:

1. Trajectory/Navigation/Mission design;
2. Flight System (including mechanical and electronic interfaces, major system trades);
3. Software Development;

4. Mission Assurance (including electronic parts, risk management, quality assurance);
5. Assembly, Test and Launch Operations; and
6. Mission Operations and End-to-End Data Flow (including flight/ground Mission Data System).

The avionics, software, and mission data system for the three missions (and other "customer" missions) will be developed in common by the X2000 First Delivery Project, based at JPL, and their numerous partners and contractors in industry, academia, and Government. Some of the electronic parts developed by X2000 will be available for use in science instruments, such as microcontrollers, memory, and power converters (see the Description Of X2000 Components Available For Use In Instrument Proposals document of the Outer Planets Program Library, available over the Internet through URL http://outerplanets.LaRC.NASA.gov/outerplanets). Each item is intended to be made available commercially and can be considered in the design of the instrument. The OP/SP Project will handle all interfaces with X2000 and will consult with PI teams as appropriate.

3.2.2.2 Relationship Between Science Teams and the Outer Planets/Solar Probe Project

The Project Scientist for Solar Probe will have overall responsibility for the coordination of the mission's science and the achievement of the mission science objectives through chairmanship of the Mission Science Team, the other members of which will be the Science Investigation Principal Investigators.

Principal Investigators and/or key members of their teams will need to be available for frequent on-line concurrent working sessions. In addition, co-location of key Science Investigation Team members may be required during high-activity periods.

All PI teams will be required to work cooperatively with the spacecraft team to resolve interfaces and requirements and to bring the total flight system capabilities (instruments plus spacecraft) into line with the constraints of the program. This will be accomplished primarily before Science Confirmation but will continue throughout the Development Phase (to launch + 30 days). If individual instruments grow such that their resource allocations are exceeded, science resources will need to be reduced either through contributions from the other instruments, descoping, or cancellation.

As with the mission design, details of the project organization and interactions will evolve over time to meet the needs of the project and mission.

3.2.3 Encounter Science Team Selection, Participation, and Management

The OP/SP development and operations environment will require that individuals selected to produce the science investigations work closely with JPL and other team members on producing investigation hardware, software, mission design, and the flight system that supports the investigations.

After launch and as the spacecraft near their science targets, NASA plans to select via a to-be-determined process a broader team of scientists to provide the expertise required to successfully conduct the observations and reduce, analyze and interpret the data. The core of the team, it is anticipated, will be those who designed the investigations during the prelaunch phase, with possible changes reflecting career moves, retirements, and the evolving knowledge base in planetary and solar science. The intent is to retain the crucial expertise needed to fulfill the science investigation, while bringing in new people who can maximize the value of the science returned from the mission.

3.2.4 Mission Assurance Requirements

OP/SP mission assurance requirements for science instruments can be found in the Instrument Mission Assurance And Safety Requirements document of the Outer Planets Program Library, available over the Internet through URL http://outerplanets.LaRC.NASA.gov/outerplanets.

3.2.5 Principal Investigator Responsibilities

Science instrument Principal Investigators (PIs) are responsible for instrument design and development, fabrication, test, calibration, and delivery of flight hardware, software, and associated support equipment, within project schedule and payload resources. The PIs are responsible for planning and operational support of instrument operation, data analysis, and overall conduct of each of their investigations.

NASA anticipates that a PI-funded instrument engineer will attend reviews and interface meetings and maintain the instrument Interface Control Document as a normal course of doing business. No sustained stay at the spacecraft integrator's site is required prior to flight unit delivery. Extended support at the spacecraft integrator's or the launch site may be necessary depending on developments during integration and test activities.

The specific responsibilities of the instrument PI include, but are not limited to, the following:

1. Developing an internal management plan and an experiment implementation plan;
2. Ensuring that the design, fabrication, development, and testing of the investigation flight elements are appropriate to the objectives of the investigation and assure qualification to the environmental and interface constraints;
3. Managing hardware and software margin to ensure successful integration and implementation of the experiment;
4. Hardware and software quality assurance and reliability and selection of parts and materials;

5. Ensuring that instrument hardware and software development meets the approved schedules and cost plans;
6. Establishing requirements, Interface Control Documents (ICDs), schedules, and transfer of funds through negotiation with the Project;
7. Ensuring the flight hardware is flight qualified and properly calibrated;
8. Participating in Project Science Group (PSG) meetings and associated working groups. PSG meetings will be held in conjunction with PI Working Group meetings every 6 months;
9. Conducting payload reviews;
10. Participating in Software Working Group (SWG) meetings, as required by the proposed science use of spacecraft computational resources and services to resolve requirements, process issues, and interface issues and to resolve resource allocations and operational timelines;
11. Supporting payload integration and system test procedure development and maintenance and payload hardware and software integration;
12. Participating in flight system tests and integrated end-to-end ground system tests and operation of any payload-unique Ground Support Equipment (GSE) in these tests;
13. Supporting definition of mission database contents, including, but not limited to, flight rules and constraints, sequences, payload telemetry, and commands;
14. Supporting integrated mission data/sequence development and flight software integration;
15. Supporting launch site operations planning, including safety, and launch site system tests at Kennedy Space Center/Cape Canaveral Air Force Station;
16. Planning and executing mission operations;
17. Ensuring that the reduction, analysis, reporting, and archiving of the results of the investigation meet with the highest scientific standards consistent with budgetary and other recognized constraints; and
18. Preparing, certifying, and releasing a final data product (to NSSDC) within six months or less of data receipt on the ground.

3.3 Deliverables

3.3.1 General

The activities in support of deliveries by the instrument Principal Investigator to the Project include, but are not limited to, the following:

1. Sign a Memorandum of Agreement with the Project that documents resource allocations;
2. Provide and maintain required documentation, including ICDs (see Sections 3.3.3 and 3.5.4);
3. Support development and maintenance of ICDs;
4. Provide monthly Technical Progress Reports and monthly Financial Management Reports;

5. Deliver flight-qualified hardware to the flight system integrator with suitable shipping containers and any protective covers required;
6. Deliver to the flight system integrator one of the following items: a) an Engineering Model, b) a Protoflight unit, or c) a payload mechanical fit-check model and payload data interface simulator (this unit is to allow testing of the transfer of command and telemetry data with the spacecraft bus and a mechanical fit check between the instruments and the spacecraft);
7. Provide necessary payload-unique GSE for stand-alone integration and launch operations;
8. Provide payload unit history logbooks including power-on time log;
9. Deliver investigation flight software to be resident in the spacecraft flight computer (see Section 3.3.3);
10. Provide timely information to establish and maintain controlled baselines for software interfaces, shared computational resources, mission data, and mission operations timelines and sequences; and
11. Archival science data products.

3.3.2 Hardware Delivery

The payload data interface/mass simulator, Engineering Model, or Protoflight unit must be delivered to the flight system integrator's site on or before 15 months before launch. The science payload flight units must be delivered on or before 12 months before launch. Payload flight units must be accompanied by all ground support equipment needed to support system test. Unit history logbooks shall accompany the flight hardware. Payload flight units must be fully qualified and calibrated before delivery; instruments will not be returned again to the PI.

3.3.3 Software

The OP/SP Software Management Plan will specify requirements on software documentation, testing, source materials, reviews, and metrics.

3.3.3.1 Software Documentation - Software Interface Control Document (ICD)

Initial definition of operational timeline requirements and related resource demands (characterized by peak and typical parameters) will be negotiated in compliance with resource usage constraints placed on the science payload by the Project and documented in an Initial Software ICD for:

1. Volatile and nonvolatile memory;
2. Observational activity and data processing algorithm frequency and duty cycle;
3. Storage demands with storage durations; and
4. I/O requirements for all classes (data bus bandwidth, command/telemetry bandwidth) including best available information on compliance with protocol standards or any unique data transfer methods.

Updated information for all items in the Initial Software ICD, with projections of final commitments for all resource demands, plus protocol compliance for all transactions using the spacecraft C&DH, including behavioral characteristics of timing where it is relevant to correct operations of the science payload/mission, is due with the Update Software ICD.

The committed baseline for all elements of the Software ICD is the third delivery, due with the Final Software ICD.

3.3.3.2 Software Documentation - Other

Requirements, design, build, test, and evaluation information that provides insight into the software implementation should be provided as they become available, in accordance with the PI's normal development plan.

3.3.3.3 Software Test: Required Evaluation Procedures

Software test procedures are required and are subject to approval. The fidelity of the procedure and level of approval corresponds to the potential risks involved in the procedure. Generally, as the software testing is done in primarily a simulation and Engineering Development Unit (EDU) environment, the risk is minimal, requiring approval from only the cognizant personnel for the item under evaluation and Spacecraft Test Laboratory (STL) operations. Circumstances that may require further approvals include:

1. Use of flight hardware in the configuration;
2. Requirements for special interfaces, either hardware or software, that may require test setup and verification; and
3. Exclusive operations or continuous operations that produce resource conflicts not reconcilable among other parties.

3.3.3.4 Software Source Materials

The mission load (all executable spacecraft and payload flight software and data) is generated as an integrated load image, including initial/nominal values for all updatable mission data/system files. To develop the mission load, source code for compilation, materials for binding, and data/file load shall be provided in a timely fashion to support software development integration in the Spacecraft Test Laboratory, assembly and integration tests during science payload integration, and mission readiness tests at the launch site. The Final Software Baseline Delivery for launch is scheduled at the time of flight hardware delivery, prior to the start of science integration for final build and characterization of the launch configuration load image. Other postlaunch flight software updates are expected.

3.4 Payload Reviews

The payload PI(s) will be expected to attend the spacecraft Preliminary Design Review (PDR) and Critical Design Review (CDR), ground system reviews, and any informal reviews scheduled by integrated development teams with payload participation requiring the PI rather than the instrument engineer.

Each instrument PI will host a Preliminary Interface Requirements and Design Review (PIRDR) for their investigation. The PIRDR is scheduled as early as possible after the completion of the Functional Requirements Document (FRD)/Experiment Implementation Plan (EIP). Topics include: discussion of the EIP, discussion of the FRD, description of interfaces, I/F verification plan, and description of the safety plan.

Likewise, each PI will host a Final Interface Requirements and Design Review (FIRDR). The FIRDR occurs prior to the mission CDR, at the completion of the payload detailed design. Topics include: status of hardware design, fabrication, test, and calibration, software design and test plans, plans for integration, description of support equipment, finalization of interfaces, command and telemetry requirements, and discussion of environmental and system tests.

Prior to delivery of the flight instrument, each instrument PI will hold a Hardware Requirements Certification Review (HRCR) to ensure that the instrument meets all of its requirements and is ready to be shipped for integration on the spacecraft.

3.5 Documentation Requirements

The following is a list and description of the minimum formal documentation that will be required from instrument PIs:

1. Memorandum of Agreement;
2. FRD/EIP/Safety (Combined);
3. GDS/MOS Requirements (Preliminary and Final);
4. ICD Major Milestones:
    - Preliminary Physical;
    - Initial Software;
    - Final Physical (start configuration control);
    - Update Software; and
    - Final Software;
5. Instrument Design Description (IDD);
6. Payload Handling Requirements List;
7. Unit History Log Books; and
8. Acceptance Data Package.

3.5.1 Memorandum of Agreement

A Memorandum of Agreement documents the investigation resource allocation (mass, power, volume and fiscal resources) between the project and each investigation PI. This is written immediately after payload selection and signed by the Project Manager, PI, and spacecraft flight system integrator designee for hardware investigations.

3.5.2 Functional Requirements Document (FRD)/Experiment Implementation Plan (EIP)/Safety Plan

Each instrument PI is responsible for writing a combined Functional Requirements Document and Experiment Implementation Plan for their investigation within 3 months of selection. Contents are negotiated with the project manager, but may be assumed to include:

1. Payload functional requirements;
2. Hardware development-and-test plans and schedule, including reliability and quality assurance plans;
3. Software development-and-test plans and schedule;
4. Cost plan for hardware and software development, fabrication, test, and calibration from selection through launch;
5. Margin management plan;
6. Post-launch cost plan for instrument operation, data analysis, and data archiving;
7. Requirements for project support;
8. Personnel and hardware safety plans;
9. Contamination control plan;
10. Calibration plans;
11. Science management and investigation plan;
12. Payload portion of range safety plan and payload safety at launch site; and
13. Fracture control plan (for Space Shuttle launched payloads).

3.5.3 Ground Data System (GDS) / Mission Operations System (MOS) Requirements

Ground Data System / Mission Operations System requirements due dates are listed below. These primarily address instrument operation requirements and flight rules.

	Solar Probe
Preliminary	9/04
Final	9/06

3.5.4 Physical Interface Control Documents (ICDs)

Physical ICDs are negotiated directly with the spacecraft engineering team in an integrated-development-team environment, with Preliminary Physical ICDs required by the spacecraft PDR and final Physical ICDs under configuration control by the spacecraft CDR. Physical

ICDs identify all payload interfaces, including, but not limited to, the volume envelope, mounting, center of mass, electrical and mechanical connections, end circuits, pyro devices, features requiring access or clearance, purge requirements, testing, facility support, view angles, clearances, etc.

3.5.5 Instrument Design Description Document (IDDD)

The final design of the payload is documented in an IDDD. The IDDD is due at the HRCR. Included in the IDDD are the parts and materials list.

3.5.6 Payload Handling Requirements

A payload handling requirements list must be supplied prior to the delivery of flight units to the spacecraft integrator. This checklist describes any special handling necessary to ensure the safety of the flight hardware.

3.5.7 Unit History Log Book

The Unit History Log Book accompanies the delivery of the flight hardware.

3.5.8 Acceptance Data Package

The Acceptance Data Package includes (but is not limited to) final drawings, documents, mass properties, qualification data, footprint drawings, final power, etc.

3.6 Key Prelaunch Delivery Dates

Activity	Due date
Contract execution	6/00
FRD/EIP	~10/02
Science Confirmation	6/03
PIRDR	~8/03
Physical ICD - preliminary	10/03
Software ICD - initial	10/03
Mission PDR	10/03
FIRDR	4/04
Software ICD - update	8/04
GDS/MOS requirements - preliminary	9/04
Physical ICD - final	9/04
IDD - preliminary	9/04
Mission CDR	9/04
SIM, EM or PFM delivery	11/05
Flight S/W for SFC - preliminary	11/05
Software ICD - final	11/05

S/W test procedures	11/05
HRCR	1/06
IDD - final	1/06
Flight Unit delivery	2/06
Unit history logs	2/06
Flight S/W for SFC - final	2/06
Payload handling requirements	2/06
Acceptance data package	2/06
GDS/MOS requirements - final	9/06
Launch	2/07

4. Discussion of Nadir Viewing Options

The prime objective of the *in situ* measurements is to find characteristic features in the ion and electron distribution functions near the Sun that provide information on how the solar wind is accelerated. Besides the bulk flow transition from sub- to supersonic acceleration, processes may include wave-particle interactions. Therefore, the plasma detector and the spacecraft must be designed to reveal such processes in the velocity distributions. However, to contain costs it must not be over-designed to obtain more information than is necessary or to look for the solar wind in unlikely directions. Wave damping and other possible acceleration mechanisms will leave characteristic traces in the plasma distribution functions, such as supra-thermal particles, ion jets, etc. Current knowledge of the solar wind indicates that these traces are found in different parallel and perpendicular temperatures, in relative velocities between hydrogen and electrons or hydrogen and other heavier ions, double streams in distributions of individual species, and nonthermal tails or heat flux as a function of the distance from the Sun. For state-of-the-art plasma instrumentation, these requirements pose no problem. However, the challenge lies in the fact that part of the key viewing range, i.e. towards the Sun, is blocked by the heat shield of the spacecraft.

The "aberration" method of viewing the solar wind has been the default option for the Solar Probe. It relies on the aberration of the plasma particles' arrival direction as seen from the rather fast-moving spacecraft (310 km/sec at perihelion) and tilting the spacecraft (when away from perihelion) so that the plasma viewing is close to the edge of the umbra. The solar wind would then be viewed around the edge of the heat shield. Ions and electrons arriving from directions within 15 degrees (or more, because the instrument is not exactly at the apex of the umbra) of the spacecraft axis (nadir) would be out of view. Whether this is a serious loss or not depends on the magnitude and direction of the bulk speed of the solar wind and the location of the diagnostic features in velocity space with respect to the bulk velocity. The aberration method will work, if the bulk speed of the solar wind is less than twice the spacecraft velocity component perpendicular to the spacecraft-Sun line. This should be the case except possible for fast streams detected over the poles. Evidence from IPS, SPARTAN and SOHO measurements cited in Section 1.3 of the State of Knowledge of the Sun--taken from the Solar Probe Science Definition Team Report document and available in the Outer Planets Program Library, available over the Internet through URL http://outerplanets.LaRC.NASA.gov/outerplanets--indicates that the speed of the fast solar wind between 4 and 20 R_S is 500 to 800 km/sec. Furthermore, the expected meridional flow of the solar wind at mid-latitudes along the trajectory and deflection by MHD waves will exacerbate this problem at times.

4.1 Solar Wind Plasma Viewing Options

To improve nadir viewing on the Solar Probe with angular and energy coverage as complete as possible, several different approaches should be considered. In principle, those are:

- Minimizing the blocked angular range dynamically by mounting the plasma spectrometer on an extendible boom that keeps the sensor at the anti-sunward tip of the umbra. This minimizes the size of the obstruction, but the nadir direction (±8 degrees at 10 R_S) is still blocked.
- Viewing of selected pieces of the distribution function through collimator tubes protruding through the heat shield. This provides a few disconnected pieces of the distribution in nadir direction, but keeps the maximum obstruction for contiguous observations.
- Providing true nadir viewing by means of an electrostatic mirror that deflects ions around the heat shield. This allows an almost full field of view, but may be limited to less than 12-18 keV/Q.

To provide better understanding of this problem, we will demonstrate how typical ion distribution functions in the solar wind are viewed by a plasma instrument in these options. Ultimately the choice will be determined by the optimum compromise between the achievable viewing and the impact on spacecraft complexity and resources.

4.2 Plasma Visualization Model

For illustration purposes, a simplified model of the solar wind with a Maxwellian distribution and a temperature anisotropy in $T_\perp$ and $T_{||}$ has been chosen. The expected bulk speeds and temperatures as a function of distance are taken from a model by Habbal et al. (1995) for solar minimum conditions. Figure 17 shows a sample solar wind distribution in color representation for the inbound leg of the trajectory over the pole at 10 R_S. Shown are a cut in the plane of the spacecraft trajectory (v_x - v_y) and a cut in a perpendicular plane (v_z - v_y) at v_X = 890 km/s. The distribution is shown in the spacecraft frame, i.e., as seen by a plasma instrument. The rest frame is indicated by an additional white cross, offset by the negative spacecraft velocity. The lines inserted in both cuts indicate the fractions of the distribution function that are not visible by a plasma sensor with fixed mounting (dashed line) and extendible mounting (full line). The obstructed portion of the distribution can be viewed up to 12-18 keV/Q using an electrostatic mirror as described in Section 4.1 above. This viewing tool can be visited on the WWW under http://satyr.msfc.nasa.gov/Solar_Probe/ to explore other custom examples.

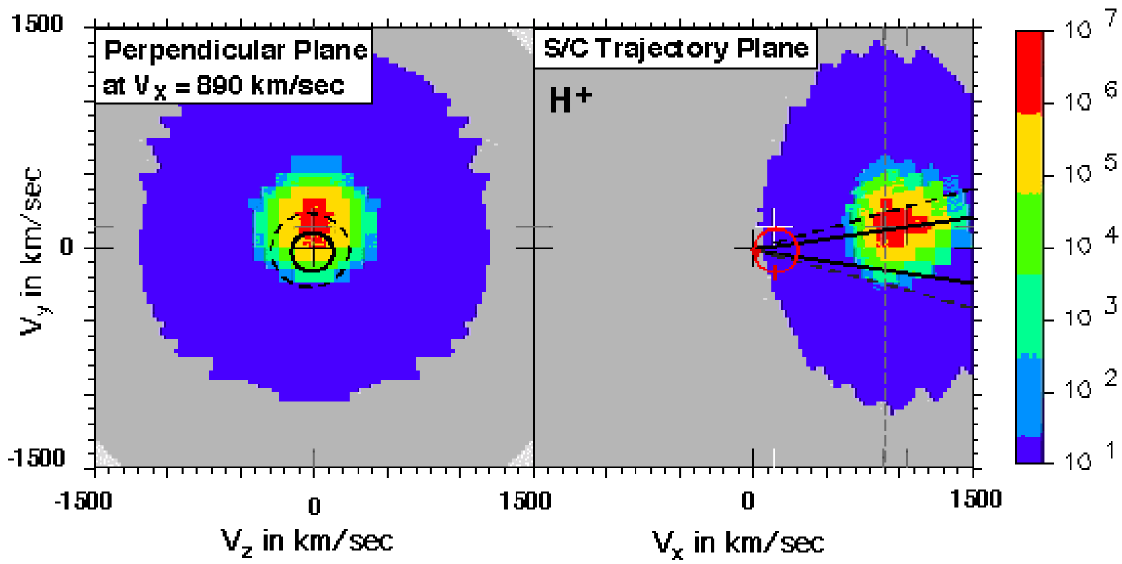

Figure 17. Sample distribution of H^+ with V_{sw} = 740 km/sec with $T_\perp = 2 \times 10^6$ K, $T_\perp/T_\parallel = 2$ and a second beam at a differential velocity of +160 km/s, displayed in the spacecraft frame at 10 R_S. Shown are two cuts: in the plane of the trajectory (right) and in a perpendicular plane at V_x = 890 km/s (left, gray line in right panel). The obstructions by the heat shield are shown as wedges and circles in both planes, respectively (dashed line: fixed mounting, full line: extendible boom mounting). An electrostatic mirror covers viewing between the dashed lines. The red circle indicates the motion of the origin of the spacecraft rest frame through velocity space along the spacecraft trajectory.

When referred to radial (v_X) and meridional (v_Y) solar coordinates in velocity space, the tip of the spacecraft velocity vector executes a counter-clockwise circle of diameter equal to the speed at perihelion along the entire trajectory and so does the 0-marker of the spacecraft frame in the right panel of Figure 17 (indicated by the circle). The speed at perihelion in a quasi-parabolic orbit is $42/\sqrt{r}$ km/sec, where r is the perihelion distance measured in AU. For Solar Probe, the speed at perihelion is 310 km/s (indicated by the white cross). With this circle, we can easily evaluate the aberration of the distribution function of the plasma at any point along the trajectory. The vector sum of the spacecraft aberration velocity and the velocity of the plasma in the solar frame provides the angle and speed in the spacecraft frame at which the plasma is viewed. In particular, it shows at each point along the trajectory what angles with respect to the spacecraft nadir the plasma instrument must view to see particular features of the distribution function and which features may be obstructed by the shield. For example, on the inbound leg the radial velocity adds to the solar wind velocity, i.e., the aberration effect on the viewing is weakened. A variable, but substantial, cone in velocity space remains invisible due to the obstruction by the shield as a function of distance from the Sun. The narrower cone in Figure 17 represents the obstructed view at 10 R_S for the variable distance (extendible/retractable) boom.

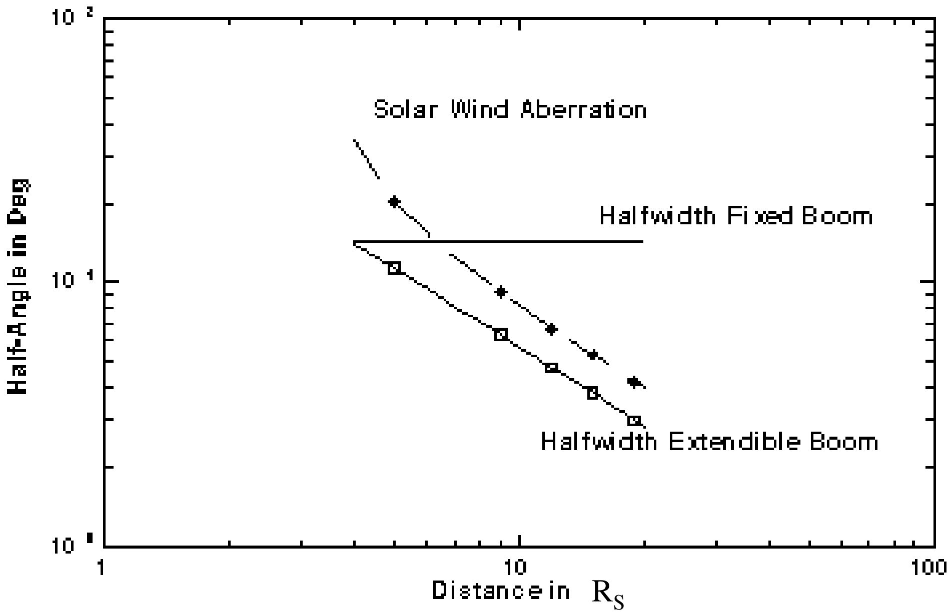

Figure 18. Variation of the half-width of the obstruction cone for an extendible boom, along with the aberration for pure radial solar wind on the inbound trajectory of the Solar Probe. It should be noted that for distances > 8 R_S (over the pole and further outbound) a meridional flow or MHD wave deflection of the solar wind of ≤ 3° can move the core of the wind distribution into or out of the obstruction. The halfwidth for fixed mounting is also shown.

The typical variation of the half width of the obstructed cone with distance from the Sun for an extendible boom is shown in Figure 18, along with the variation of the aberration angle for a radial solar wind flow of 740 km/s. While the bulk of the distribution will frequently be obstructed for a plasma instrument on a fixed mounting, the viewing of the bulk flow can possibly be preserved with a minimized obstruction using an extendible boom. However, it should be pointed out here that the figure shows a purely radial flow. A tilt of the magnetic field lines into the probe trajectory, which is very likely over the poles according to the model by Gleeson and Axford (1976), tends to move the bulk flow into the obstructed portion. Also magnetohydrodynamic waves may deflect the wind into or out of this direction. In both cases, the bulk flow could be lost and would have to be reconstructed from the peripheral view without nadir viewing. Even if the very center is not cut out, it will be difficult to reconstruct nonthermal features, such as a nonthermal tail or heat flux. Judging the substantial fraction that may be lost in and near the center of the wind distribution from the example in Figure 17, this would constitute a major reduction of the capabilities in a key scientific area for the probe for the inbound polar passage.

4.3 Assessment of nadir viewing options

Given these viewing conditions of the solar wind along the Solar Probe trajectory, we now compare the different options to cope with the requirements with regard to their capabilities and their impact on the spacecraft system. Compared are electrostatic deflection of the solar wind around the heat shield, mounting of the sensor on an extendible boom at the edge of the umbra, and viewing through apertures in the heat shield with fixed mounting of a sensor at the end of a fixed boom and at the side of the spacecraft bus. A summary of the viewing options, their capabilities and limitations, along with the impact on spacecraft resources and complexity is compiled in Table 9.

Table 9. Criteria for plasma viewing options

Performance	**Electr. Mirror**	**Extend. Boom**	**Soda Straws**	**Fixed Boom**	**Bus Mount**
Viewing	no obstruction $(E/Q)_{max} = 1.5 \cdot U_m$	minimized central obstruction	several fixed fringes of distribution	large central obstruction (30^o)	Partial view at 90^o w.r.t Sun
Cleanliness of measurement	local sublimation from instrument shield	None	secondary particles in straws?	None	None
Science Desirability	**1**	**2**	**3**	**4**	**5**
System Impact					
Mechanical	launched in position	intermittent motor torques, stability, one-time deployment after launch	alignment, vibration	None	None
Electrical	HV on boom and mirror, HV/HT cables	intermittent motor magnetic field, retractable cables	None	None	None
Thermal	heat input into bus from shield	negligible	heat input to bus	None	None
Power	HV supply and cables	noncontinuous motor	None	None	None
Mass	mirror, supply, boom, shield	boom, motor	straw structure	boom	None
System Impact	optical alignment on S/C, thermal, cable EMI	attitude control	Calibration on S/C; may change	-	-
Complexity/ Risk	high/low	medium/medium	high/low	low/low	low/low

Only an electrostatic deflection system and/or small apertures through the heat shield (coined "soda straws") will be able to provide true nadir viewing. While small apertures will add a few separated slabs of the velocity distribution at a minimum separation from each other of ≈10°, an electrostatic mirror will be able fill this void for energies up to ≈ 12 - 18 keV/charge with 5° resolution. Because the "soda straws" require a fixed mounting, contiguous coverage will not be possible in a 30° cone throughout the mission. Even the extendible boom provides more unobstructed field of view. Given the coverage and resolution, electrostatic deflection provides the preferable option to preserve viewing of a substantial portion of the solar wind distribution in particular over the solar poles.

All three improvements over the fixed sensor mounting require additional spacecraft resources and add to the complexity. The "soda straws" present the toughest challenge with their multiple breach of the shield and also with the system requirement for a final sensor calibration performed on the spacecraft. The electrostatic deflection presents a significantly higher challenge than the extendible boom because of its heat impact behind the shield and the high-voltage (HV) and high temperature (HT) integration. However, it provides the preferable science option to preserve viewing of a substantial portion of the solar wind distribution, in particular over the solar poles, which is important to the mission requirements as discussed above.

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